TRW SYSTEMS

5410-6001-R0V05

PHASE IA TASK B
FINAL TECHNICAL REPORT

VOYAGER SPACECRAFT

Volume 5
TRADEOFF ANALYSES

17 January 1966

Prepared for
California Institute of Technology
Jet Propulsion Laboratory
Pasadena, California

Under Contract Number 951113

NAS-7-100

TRW SYSTEMS GROUP
Redondo Beach, California

Comparison of Four Candidate Propulsion Systems

COMPARISON FACTORS	SOLID	LIQUID SYSTEMS		
	A Minuteman Wing VI Stage 2 (modified for Voyager) plus monopropellant midcourse	B LEM Descent Stage (modified for Voyager)	C Titan III-C Transtage (modified for Voyager)	D Custom Liquid Propulsion System
1. PROBABILITY OF SUCCESS				
Assessed value for sample mission profile (Appendix A)	0.949	0.968	0.924	* 0.969
Principal areas of uncertainty	Effects on spacecraft due to engine exhaust plume	Possible degradation in reliability due to stress corrosion of titanium propellant tanks by N_2O_4. (Minimized by reducing tank pressure during interplanetary phase)		
Developmental maturity	*Considerable flight experience; substantial modifications	Flight experience late '60s; *minimum modifications	Considerable flight experience; substantial modifications	New tankage development LEM engine
2. PERFORMANCE OF 1971 MISSION				
ΔV for orbit insertion, km/sec (based on allocated weights)	2.00 (Satisfies requirement)	2.10 (Satisfies requirement)	2.29 (Exceed desired value)	* 2.37 (Exceed desired value)
Minimum ΔV ΔV error	OK for midcourse and orbit trim Highest error for orbit insertion	*OK	OK, but jeopardized by limited propellant for auxiliary engines	*OK
3. COST ($ MILLIONS)				
Propulsion system and bus structure and mechanical subsystems (Appendix B)				
Development	47.7	* 28.1	40.3	52.9
Production—1971 mission	30.8	27.1	26.3	26.7
Total	78.5	55.2	66.6	79.6
4. FLEXIBILITY				
Propellant sources for high and low thrust	Separate	* Common	Separate	* Common
Variable ΔV for orbit insertion and accommodating mass change	No	* Yes	* Yes	* Yes
Orbit insertion ΔV for 1975-77 weight allocations, km/sec	1.11 (Sub-marginal; may be increased 5% by using Beryllium propellant)	1.20 (Acceptable)	1.30	1.35
Ability to produce greater impulse for future missions	Requires new solid motor development	*Excess propellant capacity	Excess capacity if Transtage tanks restored	Requires new design
5. EFFECTS ON SPACECRAFT DESIGN				
Flight spacecraft length	208 in.	208 in.	192 in.	208 in.
Cross section area for power	*Fixed array	*Fixed array	Deployable panels required for some solar array area	*Fixed array
Required by propulsion environment	Deployable heat shield to protect solar cells Protection for PSP Low-gain antenna abandoned or stowed	Ablative nozzle extension	Ablative nozzle extension	Ablative nozzle extension
6. HAZARD TO PLANETARY QUARANTINE	Possible ejection of contaminated solid particles after burnout			
	Possibility of meteoroid-induced rupture of propellant tanks leading to structural disintegration and ejection (Minimized by lower cross section of monopropellant tanks)	(Minimized by reducing tank pressure during cruise)		
OUTSTANDING ADVANTAGES	• Flight experience • Simplest main engine	• Probability of success • Lowest cost • Flexibility		• Probability of success • Performance
OUTSTANDING DISADVANTAGES	• Exhaust plume problem • Inflexibility • Cost of development		• Scope of modifications • Probability of success	• Cost of development • Development status

* Indicates superiority

CONTENTS

CONTENTS (Continued)

CONTENTS (Continued)

CONTENTS (Continued)

ILLUSTRATIONS

ILLUSTRATIONS (Continued)

TABLES

I. INTRODUCTION AND SUMMARY

1. DIFFERENCES BETWEEN TASK A AND TASK B STUDIES

This volume describes the tradeoff studies that were performed to provide the basis for the propulsion system selection for Voyager Task B. The tradeoff study in Task B considers the system selection from a significantly different perspective than was used in the Task A study. In Task A, the propulsion tradeoff consisted of two well-defined phases. The first phase attempted to optimize the propulsion system design through selection of component parts, operating parameters, and schematic arrangement, for a combination solid propellant retro-motor-hydrazine monopropellant midcourse velocity correction system and for a single engine liquid bipropellant system. The second phase compared spacecraft with each of these "optimum" systems. In contrast, the Task B study concentrates on the problems of applying hardware currently under development to the new Voyager requirements.

Other significant differences between the two studies are: (1) the work statement for Task B not only includes more basic alternates, but permits a greater degree of design freedom within the prescribed alternates; (2) the Task B mission description presents divergent sets of propulsion requirements for the 71-73 missions and the 75-77 missions and implies a far greater need for operational flexibility than was evident in Task A; (3) the cost of propulsion in comparison to the rest of the spacecraft is a more significant parameter in Task B than in Task A; and (4) the Task B mission description states that the orbit insertion maneuver must be performed with the flight capsule in place.

2. ORGANIZATION OF VOLUME AND CONTENTS

The volume is organized to show:

- Requirements of the mission as given by JPL in the mission description documents; the guidelines used to bound the study as assumed by TRW; and the requirements generated by vehicle design considerations and interactions between propulsion and the other spacecraft subsystems

- Criteria used for comparing and ultimately for selecting the propulsion system

- Design descriptions of the basic propulsion alternates and the rationale used to select the design options and operating parameters
- Flight spacecraft design considerations as related to each of the propulsion alternates
- Comparison of the propulsion and spacecraft options and the recommendations of the study.

3. SELECTION PROCESS

In order to reach a timely decision (so as to allow sufficient time to complete the detailed spacecraft design), it was necessary to use a selection process which quickly eliminated the least likely alternates and then converged on a recommended approach. The selection process used is shown schematically in Figure 1.

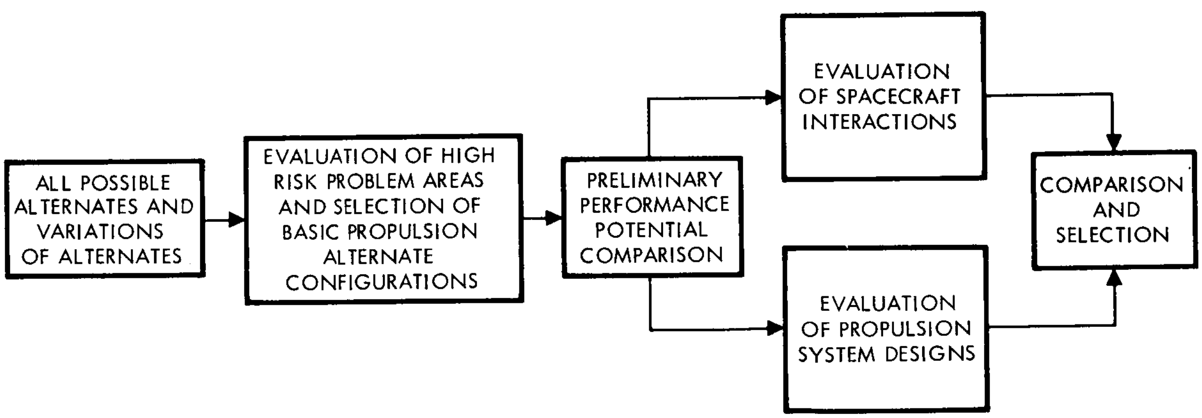


Figure 1. Propulsion System Selection Process

The first phase was an investigation of the design variables within each basic alternate system to determine if there were fundamental high risk problem areas or similarly degrading characteristics which constituted a rational basis for limiting the number of options. This phase is discussed in Section IV, which describes the basic alternates. At the conclusion of this phase, the following four basic systems were selected for additional study:

- A solid propellant retro-motor based on a modification of the Minuteman Wing VI Stage 2 motor, combined with a conventional, Mariner type, monopropellant hydrazine, midcourse velocity correction propulsion subsystem. This alternate is called the "combination system."
- A LEM descent stage propulsion system with only the minimum modifications required to adapt the stage for long term space storability and compatibility with the Voyager spacecraft interface requirements
- A Transtage propulsion system with minimum modifications required to adapt the stage for long term space storability and compatibility with the Voyager spacecraft
- A liquid propulsion system using the LEM descent stage engine and an optimized propellant feed system. This alternate is referred to as the "custom liquid system." It is included in the study to serve as a state-of-the-art standard for a liquid system optimized for Voyager, and as an upper limit against which performance and reliability comparisons of the other alternates are made.

These systems were then compared for performance potential, problem areas, cost, and reliability, both as propulsion systems and as related to an integrated spacecraft design. (Small scale layouts of complete spacecraft were made for each propulsion system. Refined spacecraft weight estimates were made, and propulsion performance capability was re-evaluated.) The performance comparison generated from the data available at this point is shown in Figures 3 and 4. Figure 2 shows orbit insertion velocity increment capability for the 1971-73 weight allocations as a function of actual midcourse velocity increment conducted. Figure 3 identifies the orbit insertion capabilities of the four configurations (after reserve for midcourse and orbit trim requirements) for the 1975-77 weight allocations, and indicates the consequent launch period available for 1975, Type I trajectories (the most critical) for four represe ntive orbits about Mars.

Each of the propulsion systems was then evaluated in consideration of the JPL competing characteristics guidelines.

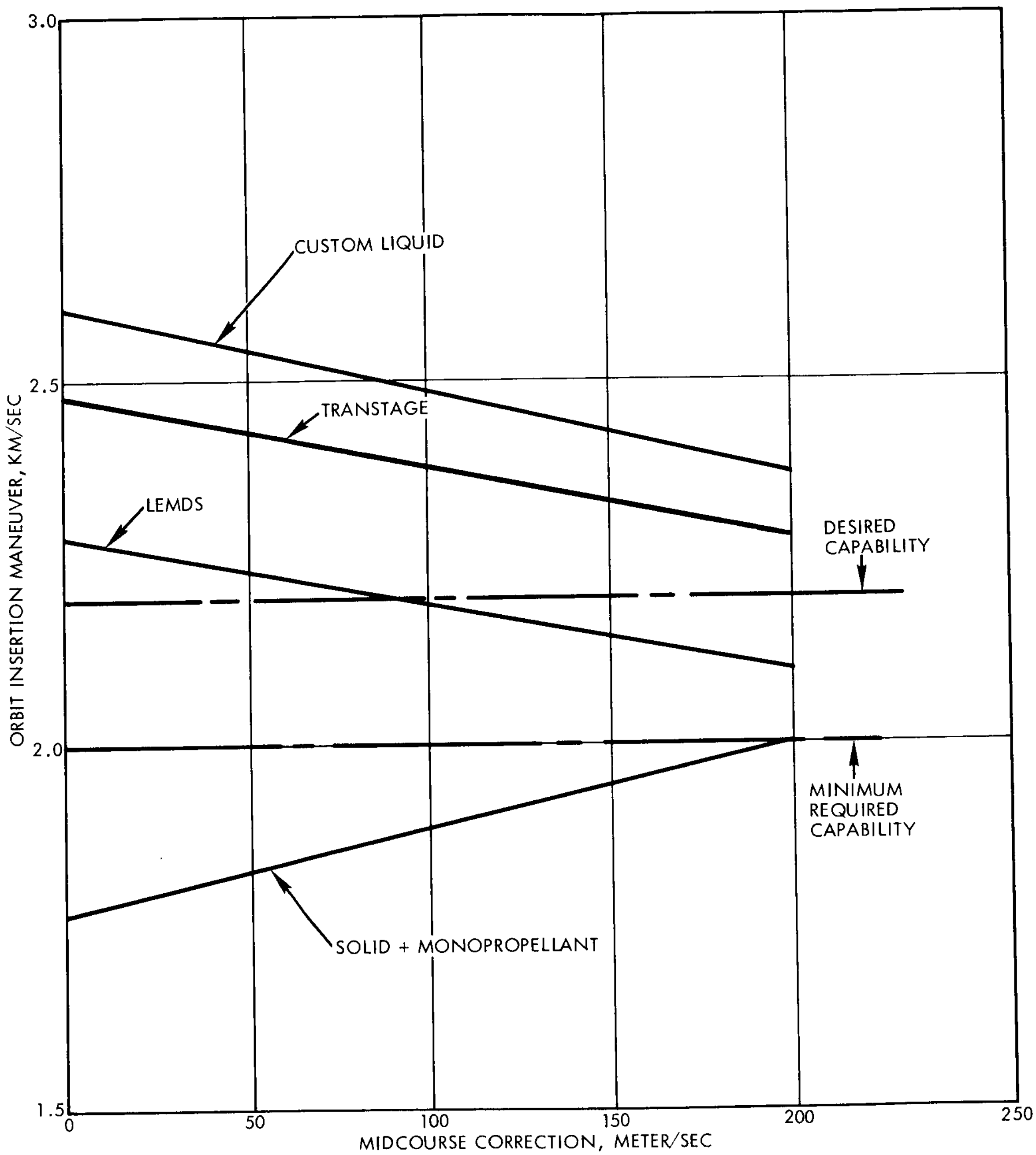


Figure 2. Preformance Comparison for 1971-73 Mission Based on Allocated Weights

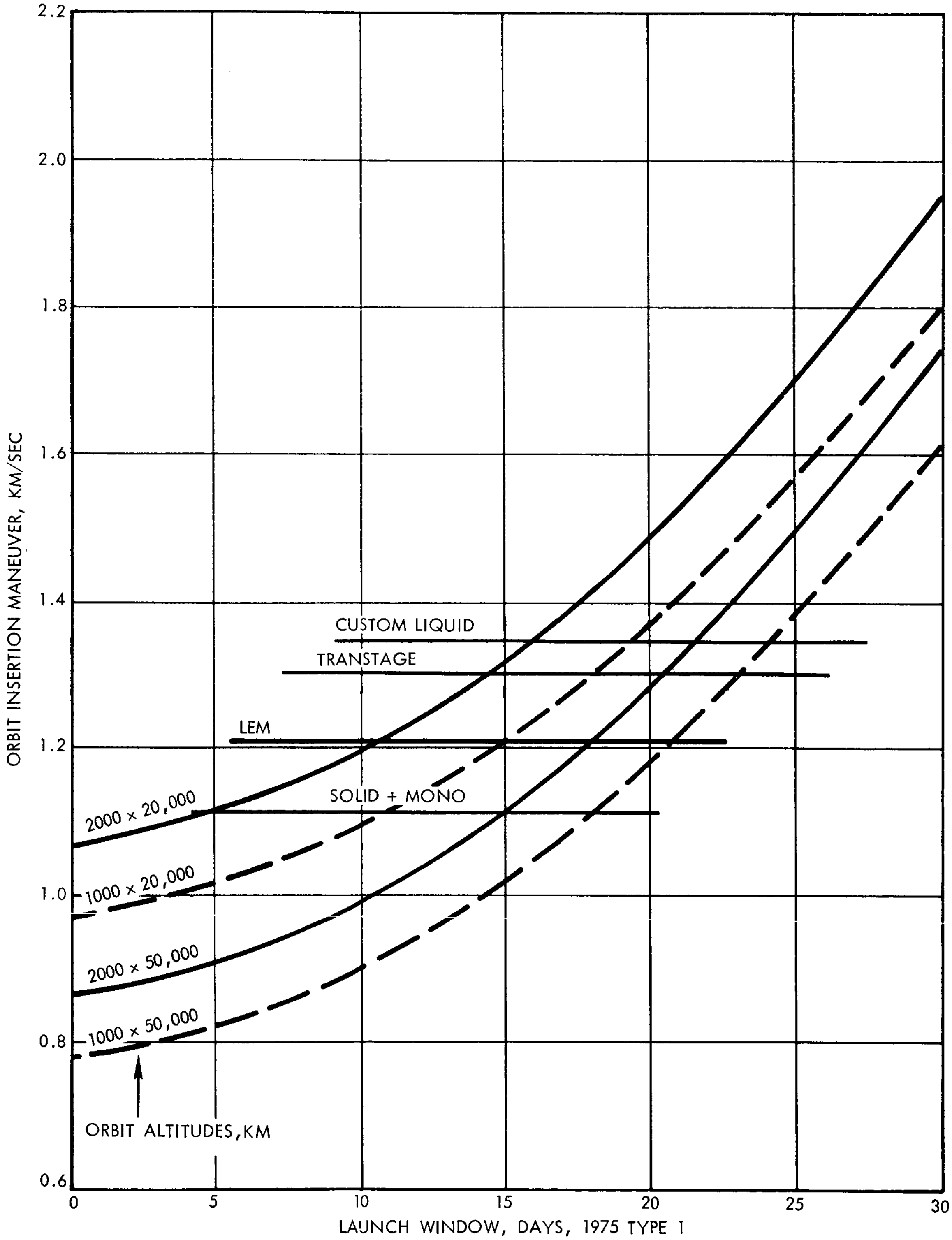


Figure 3. Performance Comparison for 1975-77 Mission Based on Allocated Weights

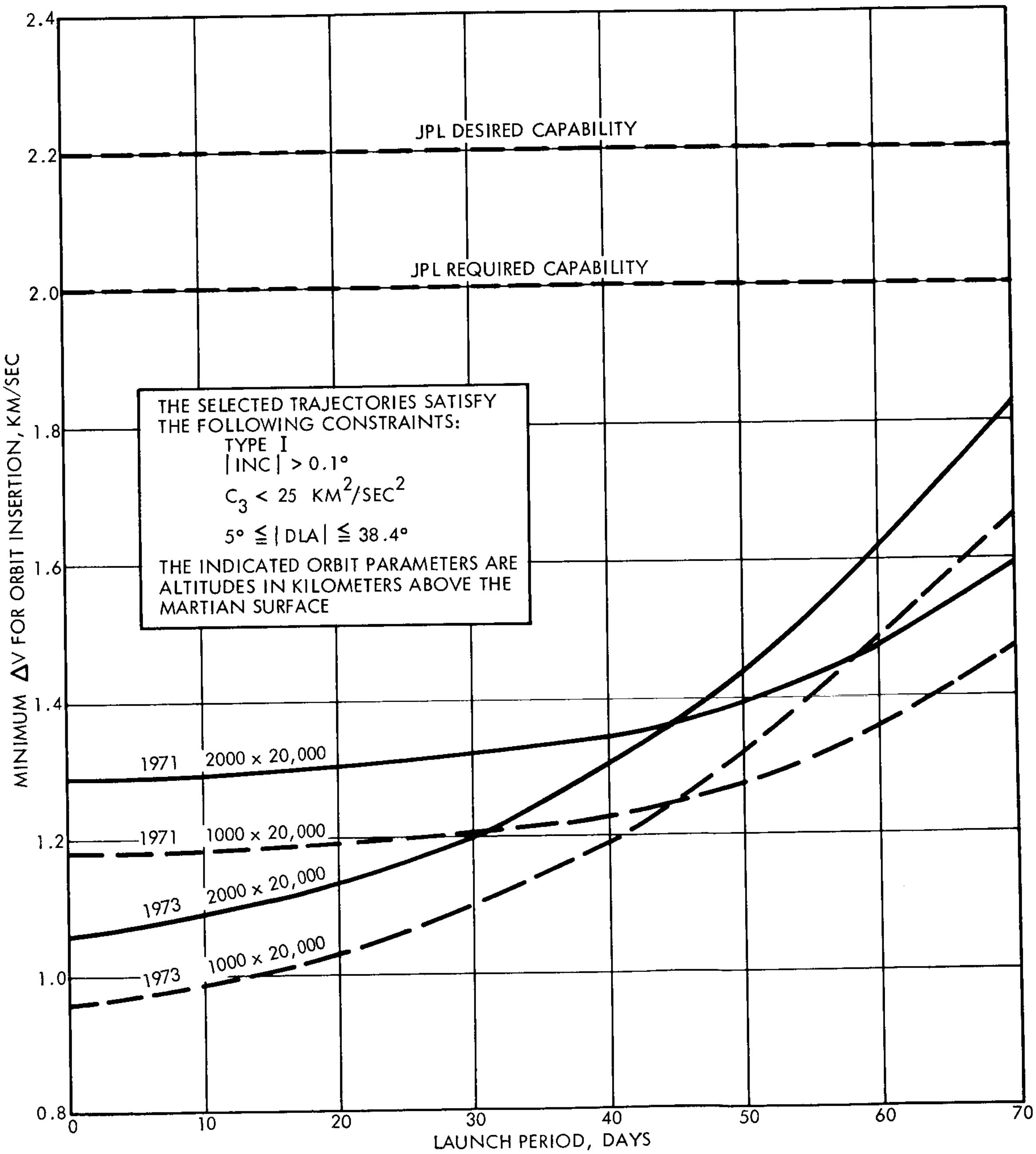


Figure 4. Velocity Increment Requirements for Martian Orbit Insertion During the 1971-73 Launch Intervals

4. RECOMMENDATION AND MAJOR CONSIDERATIONS

The results of the study indicated that the LEM descent stage, modified as indicated in Volumes 1 and 2, was the best choice. The major considerations substantiating this selection are given below.

4.1 Comparison of LEM and Transtage

As a matter of basic philosophy, the LEM stage is currently being designed and developed for a significantly longer operational life than the Transtage. Hence, considerable development, as indicated in the material furnished by JPL for the study, would be required to bring the Transtage to the level of the current LEM technology.

If a single-engine transtage were used, the structure of the stage would require major redesign, redevelopment, and requalification, and the stage length would be significantly increased. If the two-engine Transtage were used, acceleration loads to the vehicle would be increased, the problems associated with engine control during start and shutdown transients would have to be accepted. (In either event, the main engine shutoff error is too great for Voyager midcourse correction requirements. Thus, an auxiliary propulsion system is necessary for trimming maneuvers, and it might as well also be used for propellant settling for the main engine.)

In comparison, the modifications to the LEM structure to accommodate the propulsion subsystem modifications recommended in the study require substantially less development and qualification effort. Two-thrust level operation of the LEM engine provides both an efficient retro-maneuver without imposing high acceleration to the spacecraft, and precise midcourse maneuvers without requiring auxiliary trim motors.

4.2 Comparison of LEM Descent Stage and Solid Motor-Liquid Midcourse System

It was established during the study that the only advantage of a solid retro-bipropellant system as compared to a solid retro-monopropellant system for the 71-73 missions is an increase in the ΔV capability beyond the minimum requirements, and that the improvement in ΔV capability for the 75-77 missions is only about 6 percent. It was concluded that the additional development cost and the degradation to the probability of mission success are sufficient arguments to eliminate the solid retro-bipropellant system from serious consideration.

In comparing a generic liquid system to a solid-monopropellant system, the diverse requirements of the 71-73 and 75-77 flights, the mission requirements to perform the retro-maneuver with or without a capsule, and the relatively high probability of changes in the program all indicate a strong need for the flexibility and higher performance inherent in the liquid system.

Two other problems associated with the solid motor, high heat flux to the spacecraft from the exhaust plume (corresponding to exposed solar array temperatures approaching 1000°F) and high acceleration loads, were found to have practical engineering solutions. However, the solutions added a significant degree of complexity to the spacecraft which degraded the spacecraft system.

4.3 Comparison of LEM Descent Stage and Custom Liquid System

The custom liquid system exhibits the increased velocity increment capability expected to result from the weight control exercised in the design of its propellant feed system. In addition, it has a slightly higher reliability potential, because of the reduction in the number of propellant tanks. However, this development of a new design will cost substantially more than modifying the LEM descent stage for Voyager, particularly if an attempt to achieve the reliability potential is made. Furthermore, a custom design to optimize performance for the 1971 Voyager mission sacrifices other advantages. By reducing the propellant tank volume so that it does not exceed the requirement of the 1971 mission weight allocations, the possible use of the propulsion system during transit in the blowdown mode at reduced pressure (proposed in the modified LEMDS) is sacrificed. Also the flexibility to be adapted to related missions is degraded by the loss of the ability to increase propellant weight over the current allocation.

For the above reasons, and because increased 1971 performance capability over the LEMDS configuration has only secondary importance, the LEMDS configuration is preferred to the custom liquid configuration.

4.4 LEM Descent Stage Adaptability to the Voyager Mission

In Volume 2, modifications to the LEM stage are recommended to improve the long term storability and long life reliability of the stage. Once these modifications are made, and a method of operation is devised which takes full advantage of the stage's potential, the LEM stage becomes a propulsion module uniquely suitable to the Voyager mission.

The most serious of the development problems associated with any liquid propulsion system — leakage and stress corrosion — are solved by effectively isolating the high pressure gas supply system and maintaining the propellant supply pressure at low pressure during the entire interplanetary cruise phase.

It is also significant to note that the modifications recommended to adapt the LEM stage to Voyager either reduce the complexity or decrease stress levels such that the reliability is generally enhanced and development risk should not impose serious problems.

II. REQUIREMENTS

This section outlines the requirements placed on the design of the spacecraft propulsion system. It includes mission requirements imposed directly by JPL, and requirements arrived at by interpretation of other constraints.

1. TRAJECTORY REQUIREMENTS

The Voyager missions consist nominally of the following phases: launch and injection, acquisition, trajectory correction, cruise, orbital insertion, orbit correction, separation, and orbital operation. The trajectory correction, orbital insertion, and orbit correction phases are of primary interest with regard to propulsion tradeoff analyses.

1.1 Mission Sequence

Each of the propulsive phases will start with the transmission and verification of the maneuver magnitude and the turns required to obtain the desired maneuver direction. Antenna switching and reorientation may also be required if the maneuver is to be made relatively late in the mission when the omni-antenna is ineffective. Upon transmission of the enable command, the spacecraft will switch to inertial (gyro) reference and the turns will be completed. When the correct spacecraft orientation has been verified, the propulsion inhibit command will be removed and the engine will thrust until the commanded velocity increment has been obtained. The spacecraft will then be returned to celestial references via the technique used for initial acquisition of these references.

Nominally two midcourse corrections will be made with the first occurring 2 to 10 days after launch and the second 30 to 60 days prior to planetary encounter. An additional maneuver or maneuvers may be required. The orbit insertion maneuver nominally occurs during the time of the spacecraft's closest approach to the planet on its transfer trajectory. The exact timing depends on the velocity increment capability and the orientation of the desired orbit with respect to the incoming trajectory. Pre-separation orbit trim maneuvers may occur from after the first several orbits up to 10 days after insertion. An additional orbit trim maneuver or maneuvers may be desirable after the capsule/bus separation to optimize the orbit for scientific data acquisition.

1.2 Planetary Vehicle Weight Allocations

The weight allocations for the Phase IA, Task B Voyager missions were part of the specifications.[1] These figures are given in Table 1, where it may be observed that separate quotas were established for the propulsion system and bus. Since many items of structure, cabling and thermal control for example, could be arbitrarily assigned to either of these categories, TRW has adopted the position that the sum of the bus and capsule weights must be maintained within the allocation. The 1975-77 missions are an exception to this position in that it was assumed that the 1000-pound increase scheduled for these years is an increase in the spacecraft bus weight solely; i.e., the propulsion system weight would be unchanged.

Table 1. Maximum Weight Allocations

Item	1971-73	1975-77
Gross Injection Weight, lb	20,500	28,500
Flight Capsule	3,000	10,000
Spacecraft Bus and Payload	2,500	3,500
Spacecraft Propulsion	15,000	15,000

1.3 Velocity Increments Requirements

The velocity increments for the several propulsion maneuvers specified in Reference 1 are given in Table 2. In the following assessment of these requirements, indications are presented of how the capabilities inherent in these requirements would be employed to satisfy mission objectives.

1.3.1 Interplanetary Trajectory

Achievement of the 10-day arrival time separation will be provided by use of up to 150 of the 200 meter/sec total by each spacecraft to provide half of the separation ΔV. Since the 1σ requirement on the launch vehicle system for correction of the injection errors is 10 meters/sec,

(1) "Voyager 1971 Preliminary Mission Description," JPL, 15 October 1965.

Table 2. Velocity Increment Requirements

1971 and 1973 Missions	
Sum of midcourse corrections	200 meters/sec
Insertion into Martian orbit	
Required	2.0 km/sec
Desired	2.2 km/sec
Orbit trim prior to capsule separation	100 meters/sec
1975 and 1977 Missions	
Midcourse	200 meters/sec
Insertion into orbit	Maximum possible within spacecraft weight constraints
Orbit trim prior to capsule separation	100 meters/sec

adequate error correction capability will be maintained, with sufficient propellant remaining for the second and third (if necessary) corrections.

1.3.2 Orbit Insertion

The minimum (periapsis to periapsis transfer) velocity increments for injection into orbit are presented for the 1971 and 1973 launch opportunities in Figure 4. Corresponding curves for the 1975 and 1977 launch opportunities are given in Figures 5 and 6, respectively. The trajectory restrictions applied in constructing these figures are consistent with the ground rules of Reference 1. Nominal orbits are 2000 by 20,000 km altitude; however, orbits with a 1000 km periapsis have been included since this is expected to be a lower limit on this parameter. For 1975-77, orbits with an apoapsis of 50,000 km have also been included since this is a possible way of relieving the launch period limitations in these years. Figure 4 shows that a margin of 0.64 km/sec exists between the specified capability and the maximum impulsive ΔV required for a 45-day launch period. This excess capability would allow for rotation of the ellipse and/or correction of aiming point dispersions through the use of a nonoptimum insertion maneuver. Since velocity increment requirements

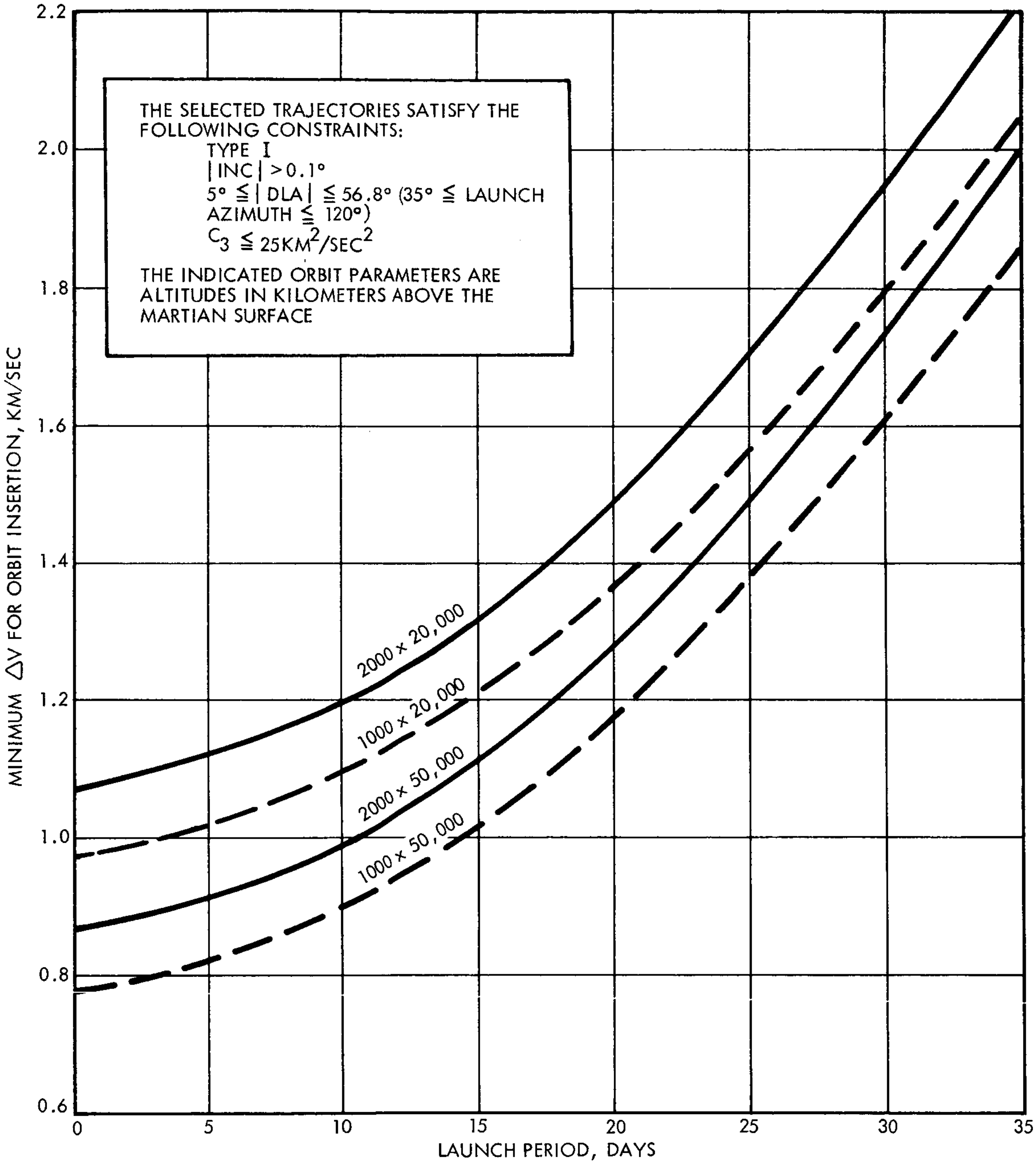


Figure 5. Velocity Increment Requirements for Martian Orbit Insertion During the 1975 Launch Interval

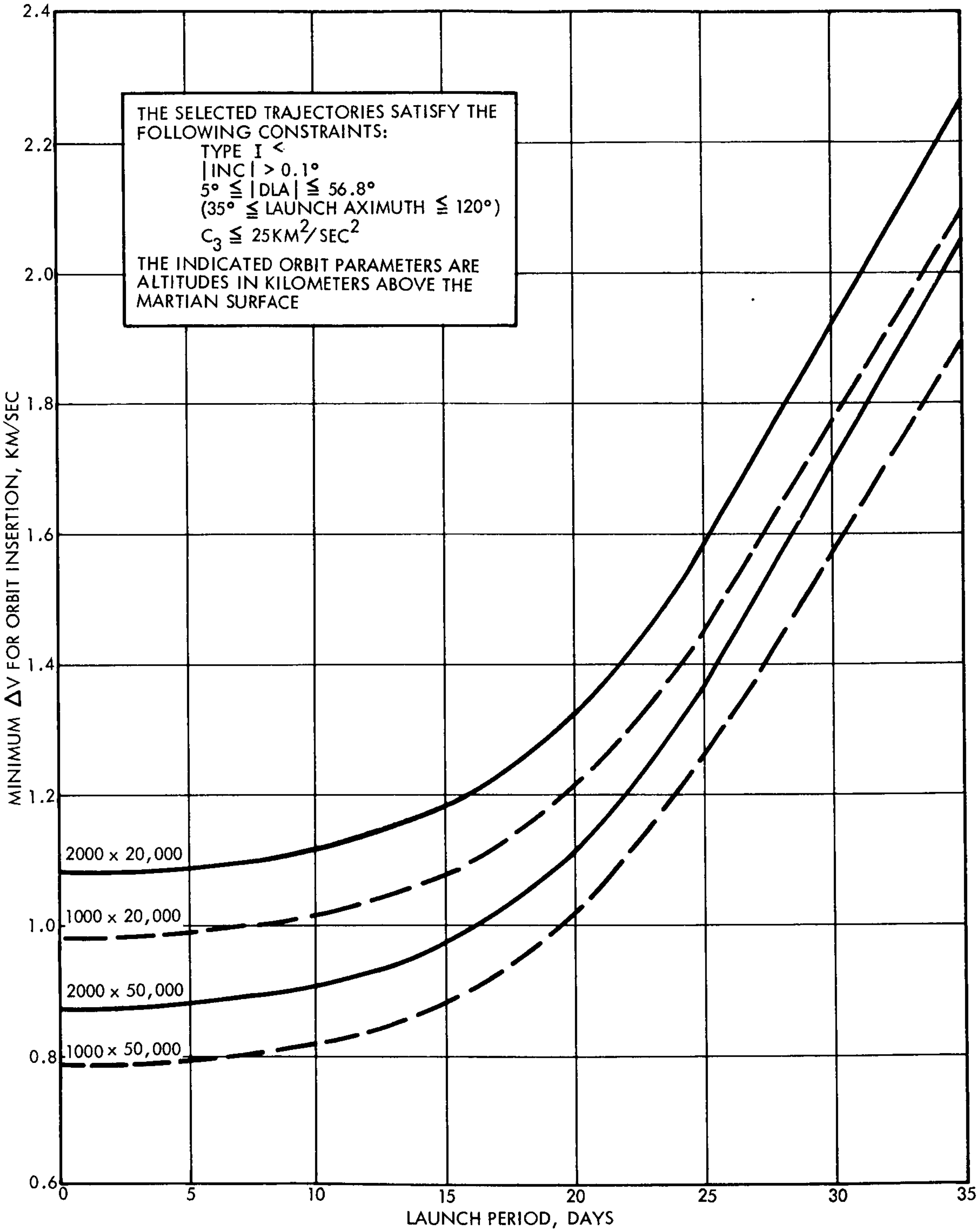


Figure 6. Velocity Increment Requirements for Martian Orbit Insertion During the 1977 Launch Interval

were not established for 1975-77, the mission capabilities for these years depend on the propulsion system capabilities.

1.3.3 Orbit Trim

Figure 7 presents the total capabilities for orbit trim as a function of the amount used prior to capsule separation. Since, as shown in Reference 2, relatively large adjustments may be made in the period and periapsis for 10:1 elliptical orbits using trim maneuvers of 50 meters/sec, the requirement for 100 meters/sec prior to separation should allow considerable flexibility in final orbit achievement. As may be seen in Figure 7, this is particularly true in 1975-77, due to the large capsule mass.

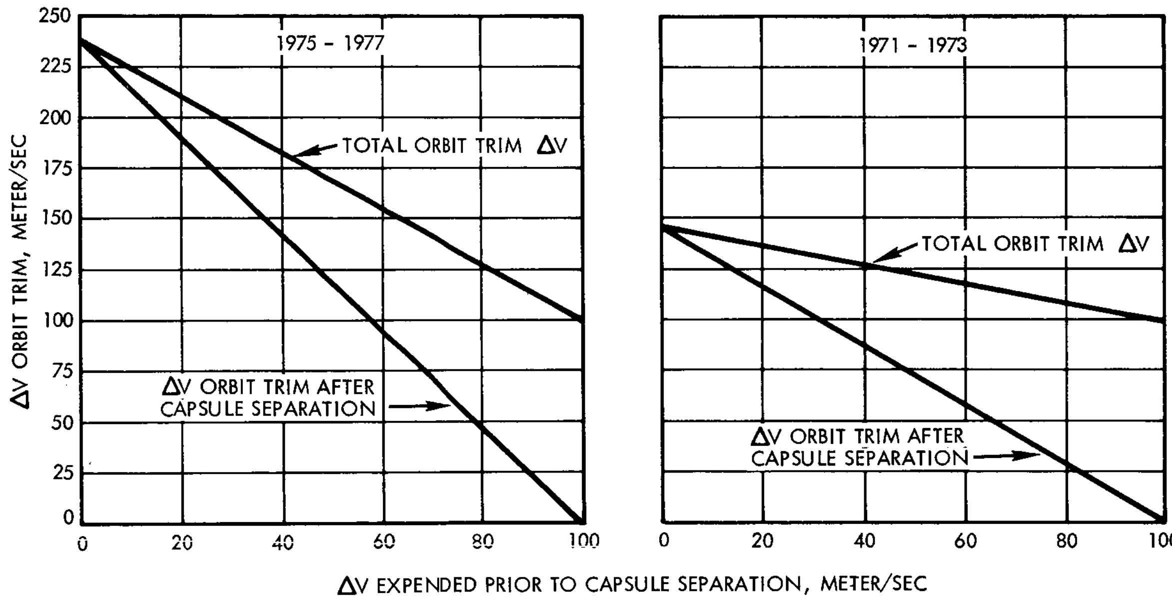


Figure 7. Orbit Trim Capabilities Prior to and Subsequent to Capsule Separation

1.3.4 Number of Starts

The minimum Voyager mission for 1971 will require four engine firings of the propulsion system: two midcourse corrections, orbit insertion, and one orbit trim maneuver. However, the nominal mission

(2) JPL EPD 281.

will require enhanced capability. It is possible that some Earth-to-Mars trajectories may require three midcourse correction maneuvers, and the number of orbit trims may be two instead of one, one before separation of the flight capsule, and one after. Additionally, it is felt desirable to have the capability of an extra start for possible use in unforeseen situations, and the ability to execute a final propulsive maneuver at the declared end of the mission if it is felt appropriate to raise the orbit altitude at that time to minimize any possibility of orbit decay for the ensuing 50-year period. This adds up to a desired capability of eight starts: three midcourse, orbit insertion, two orbit trim, one end of mission, and one spare.

1.4 Impulse Bit Requirements

The impulse bit requirement has been interpreted as follows:

- The propulsion system will be capable of performing all midcourse correction and orbit trim maneuvers with a nonproportional error of ±0.04 meter/sec (3σ). The maximum nonproportional error of the orbit insertion maneuver is taken as ±0.5 meter/sec(3σ), but the proportional error (controlled by the guidance system) is the dominant component.

- The propulsion system will have the capability of performing a minimum midcourse correction of 1.0 meter/sec.

Achievement of these objectives requires that the tail-off impulse when the engine is shut off be reproducible within approximately 10 per cent, including both errors in the thrust level and irreproducibilities in the shutoff transient.

1.5 Variations in Injected Weight

The requirement of satisfactory mission performance when the landing capsule is not attached or is separated prior to orbit insertion is best satisfied by varying the retropropulsion total impulse in flight through thrust termination. This is easily accomplished for the liquid retropropulsion motors but for a solid, an orbit which requires greater ΔV to enter must be selected. This increases the orbital error and may require a higher orbit to avoid increasing the possibility of violating the contamination constraint.

2. ENVIRONMENTAL REQUIREMENTS

The spacecraft propulsion subsystem will be designed to survive the environment imposed on it by the Voyager mission.

2.1 Launch Environment

During launch the pertinent environmental influences, which must be withstood by the spacecraft and its subsystems, are the following.

2.1.1 Static Loads

The propulsion subsystem will be designed to withstand a maximum acceleration of 7.0 g's along the booster thrust axis. The acceleration laterally will be assumed not to exceed 1.25 g.

2.1.2 Launch Vibratory and Shock Loads

The propulsion subsystem will be designed to withstand the following vibration and shock loads in addition to those that are self-induced. The random vibration environment for a payload attached directly to the shroud will be assumed to be the following omnidirectional input to the spacecraft at the attachment point to the shroud:

1) At liftoff, power spectral density peaks of 1 g^2/cps ranging from 150 to 300 cps with a 4 db/octave roll-off below 150 cps and 6db/octave roll-off above 300 cps in the envelope defining peaks; the time duration is approximately 30 seconds

2) At transonic, power spectral density peaks of 0.07 g^2/cps ranging from 300 to 600 cps with a 3 db/octave roll-off below 300 cps and 9 db/octave roll-off above 600 cps in the envelope defining peaks; the time duration is approximately 2 minutes.

The shock response due to shroud separation and spacecraft separation is approximated by an input consisting of a 200 g terminal peak sawtooth of 0.7 to 1.0 millisecond rise time.

2.2 Space Environment

For the 1971 Voyager mission, the spacecraft and its subsystem are subjected to the interplanetary space environment for the duration of the interplanetary cruise phase (transit time), which may be as low as 4 months or as long as 8 months, and to the near-Mars space environment for the duration of the orbiting phase of the mission, nominally 6 months. The propulsion system may be called upon to operate at almost any time during these periods, although nominal operating times are those indicated in paragraph 1.1. The nature of the space environment is described in detail in JPL's Voyager Environmental Predictions Document. The aspects of the environment most critical to the propulsion subsystem are summarized here:

- The exposure to the vacuum of space, and the effect on materials by the processes of outgassing and cold welding
- The conditions related to thermal balance: input (during nonoperating periods) due to solar radiation, a function of time because of varying distance from the sun, and output due to radiation to space.
- The flux of micrometeoroid particles.

3. PLANETARY QUARANTINE CONSTRAINT

The planetary quarantine constraint — the requirement that the probability be less than 10^{-4} that Mars be contaminated as the result of a single Voyager launch — does not have any quantitative interpretation for the propulsion subsystem. However, there are properties of the propulsion subsystem design and operation which do bear on the probabilities of contaminating Mars. These properties are discussed in Volume 1, Appendix E, from which we can abstract the following propulsion characteristics as being desirable:

1) The exhaust products of propulsion are preferably entirely gaseous.

2) The combustion process should subject all ejected material to a temperature-time history which guarantees sterility. (As a corollary, the amount of material ejected after the combustion process is completed should be minimized.)

3) The probability of explosion or other structural decomposition of the spacecraft initiated by micrometeoroid impact or spontaneous component failure should be minimized.

It should not be overlooked that the most significant contribution to the observance of the planetary quarantine constraint which can be made by the propulsion system lies in the achievement of a high reliability of successful operation of the propulsion maneuvers it is called upon to perform.

4. SPACECRAFT DESIGN COMPATIBILITY

The necessity that the spacecraft system design and the propulsion subsystem design be compatible is obvious, but it is not, strictly speaking, a requirement on the propulsion subsystem. Therefore, requirements are not listed here. However, for each alternate design considered, in turn, attention is paid to measures needed to enforce this compatibility. The most prominent facets of propulsion-spacecraft interaction which must be considered are these:

1) The ability of the spacecraft structure - and in particular deployed and articulated components and their drive mechanisms - to accommodate the vibration and acceleration loads created by propulsion operations.

2) The ability of the exposed spacecraft structure and components to withstand the heating effects of radiation from the propulsion exhaust plume, and the contaminating effects of particulate matter emitted in the exhaust.

3) The combined ability of the guidance and control subsystem and the propulsion subsystem to provide propulsive maneuvers to prescribed accuracy. The velocity increment produced by such a maneuver is a vector quantity, and both the magnitude and direction must be controlled. This control is exerted through functioning of both subsystems, and the accuracy required of each is itself a subject for tradeoff. See Volume I, Appendix C.

4) The compatible use by the propulsion subsystem and other spacecraft subsystems of space within the allowable vehicle envelope, and of cross section area presented in the direction of the sun.

5) The temperature control requirements of propellants and other internal propulsion subsystem components.

III. CRITERIA FOR COMPARISON

The study of the application of various types of propulsion subsystems to the Voyager mission leads to the selection of the preferred design for the Flight Spacecraft. The purpose of this section is to outline the criteria employed in the comparison of the alternate designs and the subsequent selection.

Section 1 states the JPL competing characteristics as given in the Preliminary Mission Description. Section 2 elaborates and expands on these characteristics. Although "competing characteristics" and "criteria for comparison" are not strictly synonymous, we felt that the former would serve as an appropriate basis for the latter.

1. JPL COMPETING CHARACTERISTICS

The Preliminary Mission Description states that in the event of technical conflicts affecting the following mission characteristics, the relative priorities, in decreasing order of importance, shall be as follows:

1) Probability of success
2) Performance of mission objective
3) Cost savings
4) Contributions to subsequent missions
5) Additional 1971 Mission capability.

2. ELABORATION

To use the above characteristics as criteria for the comparison of spacecraft designs based on alternate propulsion subsystems, it is necessary to elaborate on them, and to identify the various features or properties of a design which contribute to these characteristics.

Most of the pertinent features of a design (for purposes of comparison) are attributable to the propulsion subsystem per se; however, the influence of the propulsion subsystem on the design of the spacecraft system must not be ignored. The differences in spacecraft design which are required to accommodate the different propulsion alternates also contribute to the competing characteristics.

Although it is not always clear whether a feature should be classified as a propulsion subsystem feature or as a spacecraft system feature, we have attempted this division as follows:

- The first four following criteria (2.1 to 2.4) are intended to apply at the level of the propulsion subsystem, which is described and evaluated for each alternate in Section IV.

- The remaining criteria (2.5 to 2.8) are intended to apply at the level of the spacecraft system, which is described and evaluated for each alternate in Section V.

2.1 Probability of Success

Primary importance is attached to the probability of successfully accomplishing the 1971 mission objectives. The following properties of the propulsion subsystem design contribute to raising the probability of success.

- Design simplicity. Even if the reliability with which a given component performs a given function is not improved, the probability of mission success is increased if simplified design requires fewer numbers of such components, or fewer times that such functions must be performed.

- Inherent compatibility with the long-life requirement. When applied at the component level, this means not only inherent reliability, but, where appropriate, reliability of operation at the end of extended periods of inactivity during the interplanetary cruise phase of the mission.

- Developmental status and schedule risk. These are two terms describing the same phenomenon. Spacecraft hardware with a currently less mature developmental status imposes a greater risk that projected schedules will be jeopardized at some time in the future. Or, to put it another way, two alternate subsystem designs may be assessed as having equal probability of successfully performing the required functions, with the assessment based on comparable design simplicity and inherent component reliability. But we have more confidence that the design which has progressed further in development and test will actually achieve the predicted probability of success at the time of the 1971 mission. Thus we distinguish between the asymptotic reliability potentially achievable, and the actual reliability expected after a more limited development program.

- Compatibility with environments experienced by Voyager. In addition to the long-life requirement of the Voyager mission, the probability of successful operation of the propulsion subsystem depends on its compatibility with other facets of the environment. These include the launch environment (acceleration, vibration, shock, ambient pressure decrease, temperature) and the long cruise phase (zero-g, vacuum, temperature, solar radiation, energetic particles, micrometeoroids).

- Capability of failure-mode operation; redundancy. This recognizes that appropriate application of redundancy raises the probability of mission success, because the mission objectives may be achieved even if some of the components do not perform properly.

To become quantitative about the probability of mission success, it is necessary to define mission success. For the propulsion subsystem, then, we can state what functions must be performed in order for the mission to be a success. For the purpose of comparing the probabilities of success of the alternate systems described in this volume, the following functions are assumed necessary for the mission:

- Performance of three separate interplanetary trajectory corrections totaling 200 meters/sec, interspersed over a 6-month interplanetary cruise phase.

- Performance of the orbit insertion maneuver at the end of the interplanetary cruise phase.

- Performance of a single orbit trim maneuver 50 hours after orbit insertion.

This assumption is realistic in terms of actual mission life requirements, and is oversimplified in that all results are classified success or failure; no recognition is made of the possibility or value of achieving a partial success, even if not all the above functions are performed.

2.2 Performance

Performance refers to the quantitative measurement of the propulsive capability of the propulsion subsystem, as affected by the mass properties of the planetary vehicle. The following measures are appropriate:

a) Velocity increment capability. This capability may be expressed as a total for the entire mission, or it may be subtotaled separately for interplanetary correction, orbit insertion, and orbit trim maneuvers. Since the orbit insertion maneuver involves a different engine (in the Transtage and solid propulsion alternates) or a different thrust level (in the Lunar Excursion Module alternate) than the other propulsive maneuvers, it is appropriate to express the velocity increment capability as that of the orbit insertion, having reserved the specified minimum capabilities for all interplanetary corrections (200 meters/sec) and for all orbit trim maneuvers (100 meters/sec, before capsule release). This expression of velocity increment capability, based on weight allocations for the 71-73 or the 75-77 mission, is followed in this volume for the comparison of the alternate designs.*

b) Velocity increment accuracy and minimum size. The accuracy with which the velocity increment magnitude is achieved is a measure of the ability to perform the mission. This accuracy is typically composed of a proportional error and a non-proportional error. The proportional error is the dominant one for large velocity increments such as that required by orbit insertion, and the non-proportional error is the dominant one for small increments. If velocity increment accuracy is degraded, the probability is increased that an extra midcourse maneuver will be required to arrive at Mars within no more than the permitted dispersion.

   The minimum size velocity increment attainable establishes how fine an adjustment in trajectory or Martian orbit may be effected. If this minimum size is raised, it may be necessary to defer the execution

*In addition, velocity increment capabilities given in this volume (in contrast to Volume 1) carry the small penalty resulting from the assumption that the capsule canister is not ejected until after the orbit insertion and orbit trim maneuvers have been performed.

of the last interplanetary correction until an undesirably late date, resulting in degradation of the orbit determination accuracy at encounter, and in possible jeopardy to the quarantine constraint.

c) Thrust vector orientation accuracy. This has the same effect as velocity increment accuracy. Velocity increment is actually a vector quantity. Paragraph (b) pertains to the accuracy of controlling the vector magnitude, and (c) pertains to controlling the direction of the vector. It is recognized that the thrust vector orientation accuracy which is attainable is, to a large extent, outside of the realm of the propulsion subsystem. The characteristics of the guidance and control subsystem and the location (and accuracy of location) of the vehicle center of mass relative to the effective gimballing point of the thrust vector have a major influence on this accuracy.

It is seen that the measures of performance outlined above are dependent on the spacecraft system characteristics as well as the propulsion subsystem. For purposes of this outline of criteria, we consider them to be influences of the spacecraft design on propulsion subsystem performance, and not as influences of the propulsion subsystem on the spacecraft, which are considered in 2.5.

To the extent that performance meets the 1971 mission requirements, this criterion corresponds to the second competing characteristic of paragraph 1. To the extent that performance exceeds the 1971 mission requirements, it corresponds to competing characteristics 4 and 5.

2.3 Cost

The cost evaluation of the alternate propulsion subsystems must consider not only the production costs associated with the flight qualified units, but, more importantly, the cost of the development and test program necessary to establish by type acceptance testing, the adequacy of the design to perform the mission. Therefore major cost considerations are the current developmental status of the propulsion system chosen as the basis of the Voyager propulsion subsystem, and the extent of modifications proposed to adapt the system to Voyager requirements.

2.4 Flexibility

This criterion includes a number of qualities which can generally enhance competing characteristics 4 and 5 (contributions to subsequent missions and additional 1971 Mission capability). They provide values beyond the minimum requirements of the 1971 mission.

a) Commonality of propellant sources. This quality (found in the Lunar Excursion Module approach) permits a fuller utilization of all propellants aboard, and the exchanging, if desired, of orbit insertion capability for either interplanetary trajectory correction capability or orbit trim capability. A value which would accrue from this is the possibility of salvaging a mission (after a misdirected injection by the launch vehicle) requiring more than the 200 meters/sec allocated for midcourse corrections.

b) Variable versus fixed impulse for orbit insertion. This factor recognizes a major difference between the liquid and solid propellant approaches for the orbit insertion requirement. If the propulsive impulse is variable, permitting the magnitude of the velocity increment to be controlled, the orbit insertion maneuver can be controlled so as to achieve both the desired orbit plane and the desired line of apsides for the specified elliptical orbit about Mars. (If the impulse is fixed, only one of these two quantities, in general, can be controlled.) Furthermore, with variable impulse, the orbit insertion maneuver could be adjusted during the final stages of approach to Mars to compensate for deviations from the nominal trajectory which are determined to exist in the actual trajectory. Other advantages of the variable impulse for orbit insertion are:

- Accommodation of changes of mass (at orbit insertion) due to variations in spacecraft mass, variations in capsule mass, loss of capsule, or variations in propellant expended during interplanetary corrections. These variations may or may not be foreseen at the time of launch.

- Accommodation of changes in desired velocity increment to account for the variation of approach geometry and velocity with launch date.

c) Accommodation of different planetary vehicle weight combinations for different years. The same factor applies here as in the preceding paragraph, but over a time scale of years rather than within one launch opportunity. For the fixed impulse approach, different versions could be developed, one for 71-73 and one for 75-77, to satisfy the appropriate orbit insertion impulse requirements; however, this would cause an expanded development program.

d) Applicability to other missions (other planets). It is appropriate to examine the alternate propulsion-spacecraft combinations for possible use in other missions. These missions might be to Mars, but in some other mode than that of Voyager, in which a landing vehicle is launched from the spacecraft after it is established in orbit about Mars. They might also include missions to other planets.

2.5 Effect on the Spacecraft Design

The effects imposed on the spacecraft system design by the choice of propulsion subsystem are diverse, and include the following:

a) Implications on configuration and geometry; restriction on look angles. This has to do with the geometry of locating the spacecraft components. The various propulsion systems have different sizes and shapes, and are so large that the placement of other components — solar arrays, communications antennas, electronics chassis, etc., — is largely subordinate to the propulsion choice.

b) Envelope length (within shroud) required by spacecraft. The Preliminary Mission Description allocates a maximum length of 208 inches for the flight spacecraft; however, it is stated to be desirable that the actual dimension be kept as small as possible. The choice of propulsion subsystem exerts a major influence on spacecraft length.

c) Modularity of the flight spacecraft design (see 2.8)

d) Effect on spacecraft bus weight. Because of the requirements imposed by the propulsion subsystem on placement of spacecraft components, on the spacecraft structural configuration, and on the vibration and acceleration to which spacecraft components are subjected (see paragraph e), the spacecraft bus weight depends on the choice of the propulsion subsystem.

e) Environment imposed on spacecraft and capsule. The firing of the engine(s) imposes the following classes of environmental conditions on the spacecraft and capsule, and the spacecraft design must be such that the environment is withstood:

- Acceleration
- Vibration
- Shock
- Heat transfer by conduction and by radiation
- Contamination by deposition of exhaust products.

To the extent that the above effects cause the flight spacecraft dry weight to increase to the exclusion of propellant, they are considered in 2.2. To the extent that they cause variations in the spacecraft bus reliability or cost, these variations should be evaluated as part of this criterion. An example of such a variation in spacecraft reliability is the introduction of a deployable umbrella-shield over the solar array in the solid-propellant alternate to cope with the excessive radiant flux emanating from the exhaust plume.

2.6 Compatibility with Planetary Quarantine Requirements

A propulsion subsystem which meets the performance requirements with a reasonably high probability of success provides the basis for insuring that the spacecraft trajectory complies with the Mars quarantine constraint. The observance of the constraint is then the operational responsibility of the Mission Operations System. However, there are aspects of propulsion operation which lie outside the stated requirements, and should be evaluated. These include:

- The possibility that propulsion exhaust gases might carry viable microorganisms to Mars, either directly or via the capsule vehicle.
- The possibility that a rupture of a highly pressurized propellant tank will be induced by micrometeoroid penetration and, by violent disintegration, cause unsterile fragments to be ejected onto an impact trajectory. Such an event might occur either during the interplanetary cruise phase or while in orbit about Mars.

2.7 Compatibility with Prelaunch Ground Handling Sequence; Modularity

The extent to which the spacecraft bus and propulsion subsystems can be physically separable, or modularized, has an influence on the efficiency and ease of assembly and ground handling operations. In addition, the loading of liquid and solid propellants may cause differences in the sequences for assembling, encapsulating and surface sterilizing the planetary vehicle, and mating with the launch vehicle.

2.8 Testing and MOSE Requirements

The choice of propulsion subsystem has a strong influence on the way in which the spacecraft system test program is conducted. The mechanical operational support equipment (MOSE) requirements are similarly affected.

3. RELATIVE IMPORTANCE OF EACH CRITERION

It is difficult to discuss the relative importance of the various criteria in the quantitative sense of a rating formula. This is because many of the qualities of the propulsion subsystem approaches have a highly nonlinear value, when the JPL competing characteristics are applied.

For example, if we look at the propulsion impulse available for orbit insertion, we get no 1971 orbiting mission at all unless this impulse is great enough to permit insertion into some orbit from a minimum possible approach velocity. (It would have to produce a minimum velocity increment of about 900 meters/sec to allow capture in orbit about Mars.) Therefore, the utmost value must be attached to this initial impulse capability. However, JPL has established that a requirement for the performance of a useful 1971 mission is a velocity increment capability of 2000 meters/sec. But meeting this requirement is subordinated to the primary "competing characteristic" probability of success. Increasing propulsion impulse above that required to produce 2000 meters/sec (1971 mission) has fourth ranked value, if it contributes to subsequent missions (1975, 1977 orbit insertion velocity increment) or fifth-ranked value (additional 1971 mission capability) otherwise. In summary we evaluate propulsion impulse for orbit insertion as follows:

Achieve enough impulse to produce a velocity increment of 900 meters/sec (1971 weights)	Highest value
Maximize probability of success	Next value

Increase to raise velocity increment from 900 to 2000 meters/sec (1971 weights)	Next value
Minimize cost	Next value
Increase to produce velocity increment for useful 1975 or 1977 mission - say, 1200 meters/sec (1975 weights)	Next value
Any further increase	Lowest value

Other characteristics may be treated similarly. Those minimum capabilities, without which no mission is possible (such as the first 900 meters/sec orbit insertion velocity increment) comprise overriding considerations to which utmost importance is attached. Improvements in quality or capability above these minima are ranked according to the competing characteristics stated in 1 of this section. Thus the importance of a particular criterion in the selection process depends on which competing characteristic is at stake.

The application and interpretation of the criteria for selection is carried out in Section VI.

IV. PROPULSION SUBSYSTEM ALTERNATES

The work statement presents four basic alternate propulsion subsystems for evaluation for the Voyager mission. However, within each of these basic alternates there exist numerous design options. This section considers these design options and derives representative propulsion systems for these basic alternates, as well as one additional system not specified in the work statement.

The sizing of the propulsion subsystem effects the mission performance of the spacecraft, particularly in establishing the magnitude of the velocity increment available for propulsive maneuvers, and therefore it is important that in all of the subsystem alternates the motor be sized according to uniform ground rules. This is not insured by the application of the weight allocations of 2500 pounds to the spacecraft bus, and 15,000 pounds to the propulsion subsystem, because (1) it is a matter of interpretation as to whether certain components of the spacecraft system are part of the bus or part of the propulsion subsystem, (2) the nature of this interpretation varies from one propulsion alternate to another, and (3) the different propulsion alternates impose different weight requirements on the spacecraft bus, particularly in the structural subsystem.

In order to achieve the uniformity desired, the 15,000-2500-pound breakdown has been ignored in this volume, and the total of 17,500 pounds is allocated thus:

1) A spacecraft bus of common capability for all the propulsion alternates (but not necessarily the same weight.

2) All necessary structure, meteoroid protection, etc., divided between "bus structure" and "propulsion structure".

3) The remainder is available for all components of the propulsion subsystem not included in (2).

Whereas the allocations (1) and (2) amount to different weights for the different alternates, they are all arrived at by employing the same criteria. In particular, the basis for structural and mechanical weights is that used in Volume 2. Application of these ground rules should lead to proper comparative performance results. The absolute performance results are appropriate to the Task B study; however, the interpretation of these results should recognize how realistic the mission weight allocations are (for example, whether the nominal capsule weight applies rather than the allocated weight), and how accurate the aggregate weights of the spacecraft bus are.

1. COMBINATION PROPULSION SYSTEM

Combination propulsion systems are defined as those systems which use a solid propellant motor for the orbit insertion maneuver and a liquid system (either monopropellant or bipropellant) for the interplanetary trajectory (or midcourse) correction and orbit trim maneuvers. The JPL work statement for Task B directs that two basic classes of combination systems be considered:

a) A solid propellant unit for orbit insertion and a small variable impulse multiple start system for trajectory and orbit correction maneuvers. Size this unit for use in the 1971 and 1973 missions to meet the performance requirements and the weight allocation specified for the 1971 mission in the "Preliminary Voyager Mission Description," 15 October 1965.

b) A solid propellant unit for orbit insertion and a small variable impulse multiple start system for trajectory and orbit correction maneuvers. Size this unit for use in the 1975 and 1977 missions to provide the same trajectory correction and orbit trim capability as specified for the 1971 mission and to furnish the maximum practical orbit-insertion velocity within the weight allocation specified in the "Preliminary Voyager Mission Description," 15 October 1965.

Thus (a) is optimized for the 1971-73 missions (implicitly recognizing the need for a different solid motor to be used in 1975-77), and (b) is optimized for the 1975-77 missions, but applicable also to 1971-73. Figure 8 indicates several mechanizations of each of these main alternates, with a qualitative illustration of how the weight allocation to solid and liquid propellants varies. It applies whether the liquid is monopropellant or bipropellant. It can be seen that the reduction in the maximum size of the solid motor for 1975-77 is a consequence of the constant propulsion system weight allocation and the increased liquid propellant necessary to provide 200 meters/sec midcourse ΔV and 100 meters/sec orbit trim ΔV for the heavier planetary vehicle of 1975-77. This is why a solid engine optimized for 1971-73 cannot be used for 1975-77 missions as defined. Figure 8 indicates some mechanizations for these main alternates in which excess propellant capacity results in off-loading. This off-loading is not intended to apply to different launch times in the same opportunity, but merely to one launch opportunity as compared with another.

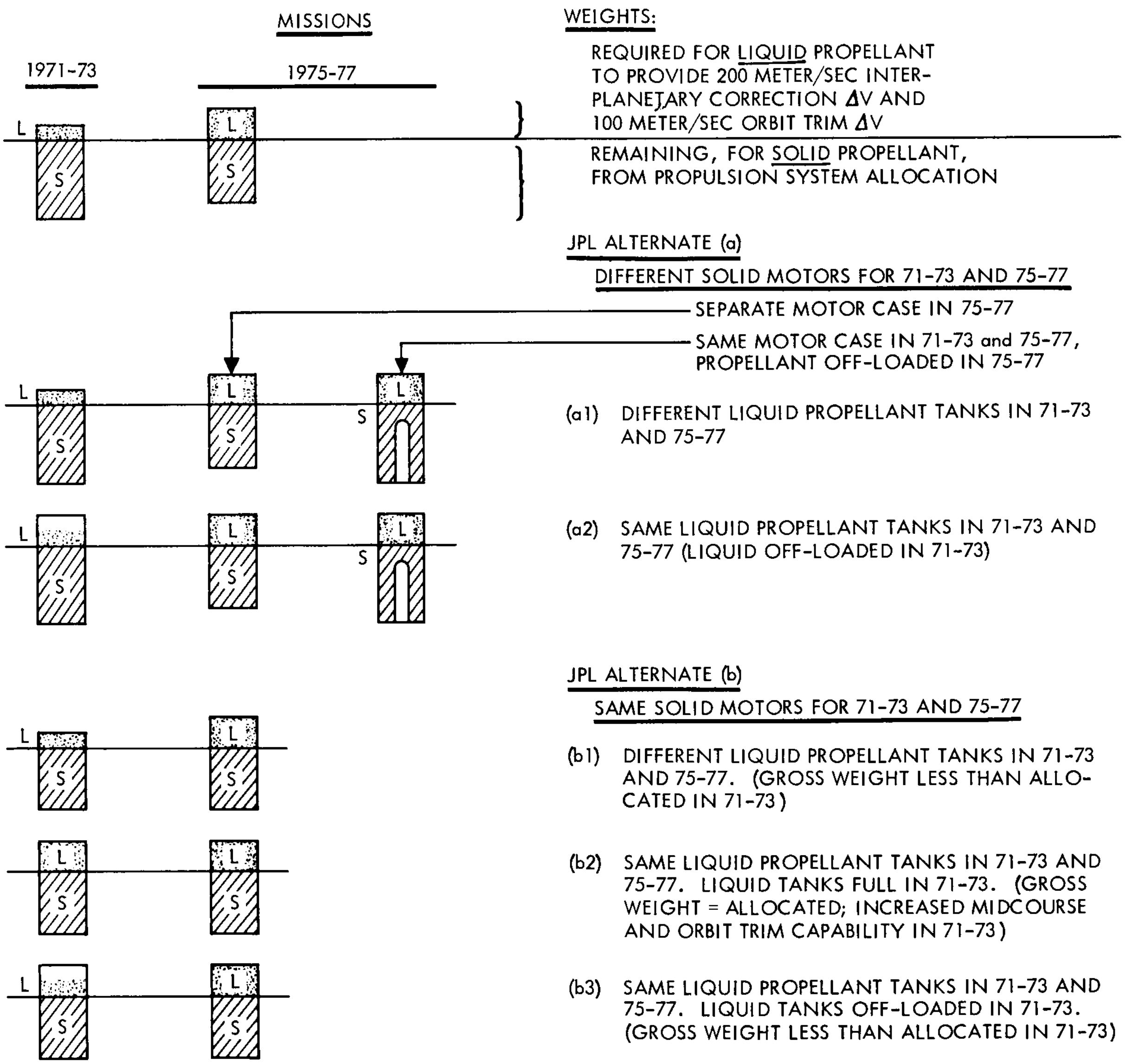


Figure 8. Alternate Solid Motor Configuration

Several of the mechanizations of Figure 8 have been analyzed for velocity increment capability, and the results are given in Table 3. Both monopropellant and bipropellant liquid systems are considered. In Table 3, the weight available for the propulsion subsystem components followed the ground rules outlined in the introduction to this section, in effect precluding full utilization of the 15,000-pound propulsion allocation of the JPL mission description. This table indicates that none of the mechanizations of alternate (b) provide enough capability to meet the 1971 mission requirement of 2.0 km/sec ΔV for orbit insertion. On this basis, alternate (a) must be employed in the consideration of a solid motor for the Voyager spacecraft—that alternate (b) fails to meet the 1971 requirements. Therefore, performance calculations for the solid motor alternate are based on the use of one motor, optimized for the 1971-73 mission and a separate motor for 1975-77.

Further, a comparison of the monopropellant and bipropellant system capabilities does not indicate a significant performance increase by using higher specific impulse propellants for the midcourse and orbit trim maneuvers. Solid motors loaded with beryllium propellants also offer some improvement (approximately 5 per cent) in ΔV capability, but are reserved by JPL input data for launch opportunities starting in 1975. (The use of a separate solid motor for 1975-77 from that of 1971-73 makes it easier to take advantage of the improved propellants.)

In summary, the following conclusions are reached for the application of combination systems to the Voyager spacecraft within the ground rules of the Task B study:

- A solid motor loaded with an aluminized propellant and a monopropellant midcourse and orbit trim system will provide adequate performance to meet the minimum 1971-73 orbit insertion ΔV requirements.

- For the 1975-77 missions an optimized solid motor using beryllium propellant, and possibly a change to a bipropellant midcourse and orbit trim system would appear to be required to provide adequate launch period to assure orbit insertion.

The foregoing considerations become academic if the 3000- and 10,000-pound allocations for flight capsule weights are superseded, and

Table 3. Combination Propulsion Systems Performance

Liquid Propellant System	Monopropellant				Bipropellant			
Mission	71-73*	71-73	71-73	75-77	71-73*	71-73	71-73	75-77
Solid motor sized for these missions:	71-73	75-77	75-77	75-77	71-73	75-77	75-77	75-77
Example of Figure 8	(a1)	(b3)	(b2)	(b1, 2, or 3)	(a1)	(b3)	(b2)	(b1, 2, or 3)
WEIGHT DISTRIBUTION								
Planetary vehicle launch weight	20,500	19,391	20,500	28,500	20,500	19,584	20,500	28,500
Solid motor gross	10,687	9,461	9,461	9,461	11,207	10,211	10,211	10,211
Solid propellant and inert expendables	9,423	8,342	8,342	8,342	9,894	8,837	8,837	8,837
Inert midcourse and orbit trim	418	600	600	600	319	483	483	483
Liquid propellant	2,081	2,016	3,125	3,125	1,660	1,576	2,492	2,492
Total propulsion exclusive of structure and contingency	13,186	12,077	13,186	13,186	13,186	12,270	13,186	13,186
Remainder of 15,000-lb propulsion allocation	1,814	2,923 (of which 1,814 is used)	1,814	1,814	1,814	2,730 (of which 1,814 is used)	1,814	1,814
Total propulsion allocation	15,000	15,000	15,000	15,000	15,000	15,000	15,000	15,000
PERFORMANCE ***								
ΔV (Orbit Insertion), km/sec	2.00	1.83	1.69	1.11**	2.10	1.91	1.78	1.16**
ΔV (Orbit Trim), m/sec	100	100	342	100	100	100	360	100

* A solid motor designed exclusively for the 1971-73 missions can not be used for 75-77 missions because insufficient weight would be available for the necessary midcourse and orbit trim propellant.

** An increase of approximately 5% can be achieved with Beryllium-loaded solid propellant.

***All systems provide 200 m/sec midcourse capability.

nominal 2000- and 8000-pound weights are used for 1971-73 and 1975-77, respectively. In that case a combination system sized for 1975-77 is suitable for all the opportunities, and a monopropellant midcourse and orbit trim system would be selected.

1.1 Solid Propellant Retro-Motor

Solid propellant motors considered for performing the orbit insertion maneuver included existing motors, modified existing motors, and an entirely new motor designed specifically for the Voyager mission. Considerations for the application of these various solid motors were based on the criteria discussed in Section III.

1.1.1 Applicability of Available Motors

A survey was conducted to determine the availability of motors suitable for the Voyager mission. The results indicated that very few existing solid propellant motors have the desired performance and design characteristics. Motors of the size and total impulse required were primarily available from ballistic missile and space research programs, e.g., Minuteman, Polaris, Poseidon, etc. However, nearly all of these motors are either too large or too small and therefore would need some degree of modification for Voyager. For example, the X260 Polaris A3 second stage is currently in operational usage with the U. S. Navy, but develops approximately 25 per cent less than the desired total impulse to provide 2.2 km/sec ΔV during the retro-maneuver. On the other hand, the second stage Poseidon C3, currently being developed by the Hercules Powder Company, is larger than necessary and would, therefore, require modification for use as the Voyager retro-motor. Results of this investigation established that existing Minuteman Wing V and Wing VI Stage II motors, currently in operational usage, should offer the greatest potential for Voyager application.

One other possibility which warranted consideration was the utilization of multiple motors to achieve the required performance. It was determined that there are several motors which, when fired in multiples of two, three, or four, could develop approximately the required total impulse. However, the use of multiple motors has inherent disadvantages related to reliability, spacecraft installation, and performance characteristics, e.g., thrust variations during tailoff could induce a detrimental tumbling action which would be difficult to correct. Further, of the motors which were considered to be applicable (with minor modification) from the standpoint of performance, none could meet weight and envelope constraints for the multiple installation. Based on these considerations, it was decided that further effort on the multiple solid motor approach be discontinued.

The Minuteman Wing V Stage 2 motor* has been fully developed for military operations, but has not been qualified for space missions. This motor is 44.3 inches in diameter and has an overall length of 155.6 inches. It has four swivel nozzles, with each nozzle movable in one plane only and capable of being stopped at any intermediate position to provide thrust vector deflection within a range of ±6 degrees. These nozzles are individually controlled by a hydraulic actuation system. The chamber is forged and machined from 6Al-4V titanium alloy and is insulated internally with prefabricated asbestos and silica-loaded nitrite rubber. A case-bonded bipropellant with a four-point star grain configuration is used to provide the desired neutral burning curve. The ignition system utilizes a propellant type igniter with an electromechanical safety-arming device. External insulation is provided to the chamber wall, aft closure, and nozzle for protection from aerodynamic heating due to recirculation of exhaust gases.

This solid propellant rocket motor very nearly meets the retro-maneuver performance requirements of the 1971-73 Voyager missions with either a monopropellant or a bipropellant midcourse and orbit trim propulsion system. However, the utilization of this motor for Voyager has certain inherent disadvantages which may necessitate modifications to existing units. Further, this motor, which is produced by the Aerojet-General Corporation, has not been in production for several years. Thus, new tooling and other operations for initiation of a fabrication program

*Detail design, performance and reliability data for this motor, as well as for the Minuteman Wing VI and Poseidon motors, are not presented in this report due to security classification.

would be required, which would dilute the cost and reliability advantages associated with utilizing the motor. (It is understood that a program for field replacement of these motors with Wing VI motors is currently being conducted, but if these surplus motors were to be considered for Voyager, they would have a storage lifetime of approximately 10 years by the time of usage, which would introduce serious reliability factors relative to the Voyager mission. One possibility for the relief of this condition might be to dissolve the propellant from the motor and then reload, but this would introduce additional complications and would not entirely resolve the storage reliability program.)

The primary modifications foreseen to make this motor useful for Voyager are those required to assure its successful operation in space. Investigations and appropriate action would be required in the following areas:

1) Necessity of providing some minimum gas pressure within the grain cavity to prevent either physical or ballistic degradation of the propellant, liner and insulation, and bonds

2) Standardized S and A and ignition squib to assure its compatibility with space environment

3) Hinged nozzle material compatibility for cold weld problems

4) Nozzle control unit compatibility with space environment, and means of modifying or sealing unit to assure compatibility with space environment

5) External insulation requirement. (It is noted that the spacecraft will provide thermal control for the motor. The external insulation on the cylindrical portion of the case is not needed. Motor base insulation is required and may not be removed.)

The Minuteman Wing VI Stage 2 motor, also manufactured by Aerojet, is larger than the Wing V motor and utilizes a single submerged nozzle with a liquid secondary injection TVC system. Thrust vector control in the pitch and yaw axis is accomplished by injecting liquid through a valve located in each of the four quadrants on the nozzle. The liquid is contained in a toroidal tank mounted at the base of the nozzle; a gas generator supplies pressure for expelling the liquid through valves on the nozzle. Roll control is accomplished by exhausting gases, created by a gas generator, through nozzle assemblies mounted on the aft skirt. The motor has an overall length of 162.3 inches, a diameter of 52.0 inches, and is loaded with 13,739 pounds of ANB-3066 high-energy propellant. Other features and characteristics of the Wing VI motor are similar to the Wing V motor. The Wing VI motor was considered unsuitable for application to Voyager, in the unmodified configuration, because, in addition to the need for modifications to assure its space storability and other modifications, the gross weight of the loaded motor is in excess of 15,000 pounds, which would violate spacecraft weight limitations imposed by the JPL requirements.

1.1.2 Applicability of Modified Available Motors

Based on the foregoing it appears that, with appropriate modifications, the Minuteman Wing VI motor could be adapted to the Voyager mission requirements. In addition, information recently received indicates that the Second Stage Poseidon C3 motor could also be modified to meet these requirements. The Poseidon C3 is slightly heavier, develops approximately 10 per cent greater total impulse, and is shorted and wider than the Minuteman Wing VI motor. However, the C3 is currently under development for the U. S. Navy and therefore does not have a proven history of successful flight experience, and its availability for the Voyager program is uncertain. Thus, for purposes of the study, the modified Wing VI motor was selected for comparison with the other propulsion concepts.

Results of preliminary TRW studies indicated that Wing VI modifications required to meet Voyager mission requirements include reduction of propellant weight, removal of unneeded capabilities, and adaption to space environment. Details of these modifications are:

	Modification	Reason
1.	Remove propellant	Removing approximately 20 inches from the cylindrical section of this motor will remove sufficient propellant and provide a regressive thrust versus time curve with minimum modification to the grain.
2.	Remove roll control subsystem	Voyager roll control will be accomplished by a cold gas system. The Minuteman system is a "bolt-on" system which can be removed as a unit.

	Modification	Reason
3.	Revise LITVC pressurization system	Minuteman LITVC system is pressurized by a solid propellant gas generator which has a predetermined gas generation schedule. This generator is not directly suitable for Voyager and should be replaced with cold gas pressure source.
4.	New nozzle	To maintain design chamber pressure and consequently burning duration, the nozzle throat area must be reduced. Also, as nozzle length or exit diameter is not a critical constraint, the expansion ratio can be increased to achieve higher specific impulse.
5.	Remove external insulation	The spacecraft will provide thermal control for the motor and therefore, external insulation on the cylindrical portion of the case is not needed.

In addition to the above modifications, investigation and appropriate action will be required in the general areas described previously for the Wing V motor.

The Aerojet-General Corporation has submitted preliminary design data based on modifications (1) through (5), above. Figure 9 shows the design presented, which is based on providing a velocity increment of 7218 ft/sec to a payload weight of 8750 pounds. Other pertinent performance and design data for the modified Wing VI motor are:

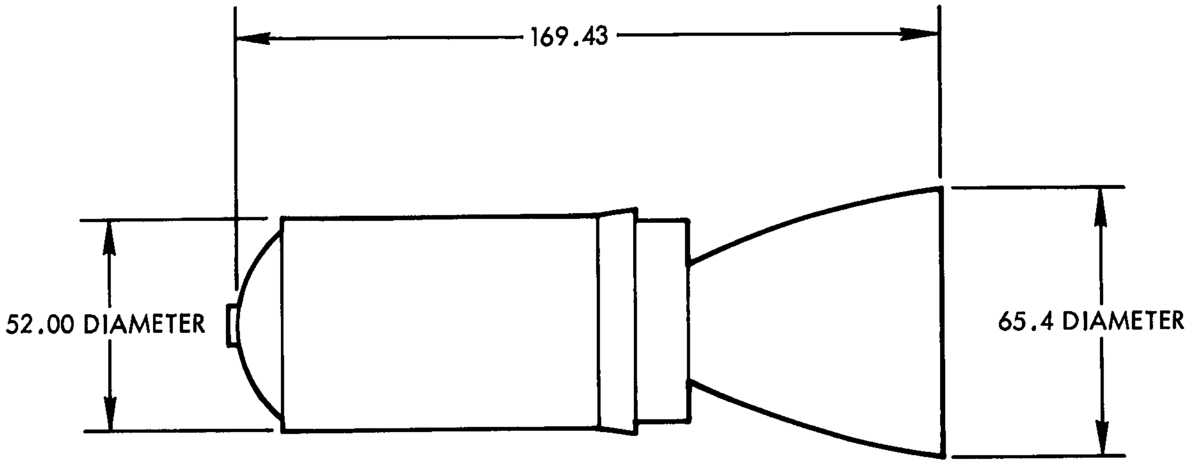
a) Gross motor weight is less than 12,500 pounds

b) Nozzle throat diameter is reduced from 9.6 to 7.8 inches and expansion ratio is increased to 70:1

c) A cold gas generator is provided for the LITVC and roll control provisions removed

d) 78 pounds of unnecessary external insulation were removed.

Additionally, it was indicated that the maximum 3 σ velocity increment variability would be 0.317 per cent and that the development cost and development time would be 50 and 60 per cent, respectively, of that required for a custom motor configuration.



DESIGN FEATURES:

CASE MATERIAL:	TITANIUM
PROPELLANT MATERIAL:	ANB-3066
THRUST VECTOR CONTROL:	LIQUID INJECTION
NOZZLE EXPANSION RATIO:	70:1

Figure 9. Aerojet-General Corporation Proposed Orbit Insertion Motor for Voyager

Flight spacecraft integration studies with this modified Wing VI motor configuration were then conducted, the results of which are discussed in 1 of Section V.

1.1.3 Custom Motor Considerations

A third possibility for a solid retro-motor concept for Voyager is the development of an entirely new motor employing advanced solid propellant technology. This concept would result in an optimized motor but would be more costly than using either existing or modified existing motors. Various solid motor manufacturers are currently engaged in research and development activities relating to advanced concepts, a few of which are:

- Long term space storability of components and materials
- Sterilization of propellants and other materials
- Higher specific impulse propellants (beryllium additives, etc.)
- Improved thrust vector control techniques.

Other areas whereby significant advances have been made include: lightweight chambers for high mass ratio, erosion resistant nozzles, and improved reliability characteristics of ignition systems.

TRW evaluated the various features and characteristics in the Task A study and determined that the solid motors had three basic problem areas:

1) Burn Time. In order to achieve desirable acceleration characteristics (3 g's maximum), a relatively long propellant burn duration (90 to 100 seconds minimum) is required. Propellants with higher performance characteristics would require additional development and testing to achieve burning rates consistent with this requirement.

2) Space Storability. No previous flight experience exists with respect to long term storability of solid rocket motors in space. However, the various motor manufacturers recognize this potential problem area and have been conducting materials and components tests to evaluate the effects of the space environment. Generally, propellant outgassing and Fiberglass case permeation present the most serious problems associated with long term storability.

3) Liquid Injection Thrust Vector Control System. The development of the LITVC introduces performance and design problems which, while not insurmountable, will require extensive component and systems testing to ensure compliance with mission objectives.

Generally, the same problem areas would be applicable to scaled up versions of the proposed designs should the custom solid configuration approach be selected to meet Task B requirements.

Aerojet has performed preliminary design studies comparing the modified Wing VI motor with an optimum motor for a payload weight of 8750 pounds. Results of these studies show that, for the same performance requirements, the optimum configuration would be shorter (90.9 inches), wider (82.2 inches), would weigh approximately 500 pounds less, and would have a slightly higher mass fraction and longer burn duration.

The results of studies* performed by the Thiokol Chemical Corporation have been made available to TRW Systems. These studies were performed to define typical performance and design characteristics of motors sized to the 1971-73 and 1975-77 Voyager mission requirements. It should be noted that Thiokol did not have a clear understanding of the mission or the JPL weight constraints. Hence, the performance values are inconsistent with those actually obtainable. A summary of significant information presented that:

- An appreciable weight saving would be realized by using fiberglass reinforced plastic rather than titanium as a case material. The 1971-73 Fiberglass motor weighs approximately 350 pounds less than the titanium motor.
- For the 1971-73 mission, using an aluminized propellant formulation, a motor optimized for minimum weight to provide a velocity increment of 2.2 km/sec to a payload of 6235 pounds would weigh initially 9052 pounds, would have an overall length of 114.6 inches, a diameter of 52 inches, and a mass fraction of 0.893. The maximum payload acceleration with this motor will be 3.0 g.
- For the 1975-77 mission, using a beryllium propellant formulation, the optimum motor (based on estimated allowable motor weight limitations and on payload weight of 13,000 pounds) would have an overall length of 140.6 inches, a diameter of 52.0 inches, and a propellant weight of 10,066 pounds for a mass fraction of 0.91. A velocity increment of 1614 meters/sec would be provided at a maximum acceleration of 3.2 g. A motor of the same initial gross weight, but using an aluminum propellant, would provide a velocity increment of 1539 meters/sec to the same payload.
- Offloaded configurations based on meeting the 1971-73 performance requirements using the larger size 1975-77 motor hardware were considered. Results would favor the aluminized propellant motor design for the 1975-77 mission because the variation of ballistic properties between aluminum and beryllium would result in a maximum acceleration of 4.6 g for the offloaded 1975-77 beryllium motor, when loaded with aluminum propellant.

---

*Thiokol Technical Report V-65-10, Vol. I "Technical Analysis of the Voyager Solid Orbit Injection Motor," 10 December 1965.

- A development schedule is feasible for motors to meet 1971-73 requirements using an aluminized propellant (TP-H1109) and for motors to meet 1975-77 requirements using a beryllium propellant (TP-H1110), including the necessary space storage qualification tests.

- A preliminary predicted inherent reliability of 0.9939 and a recommended design objective reliability of 0.996 or 0.997 was indicated. (These values were not substantiated.)

- For the total program, a budgetary cost estimate of approximately $7.5 million for 1971-73 motors and $10.8 million for 1975-77 motors was indicated. (These values are not considered to be realistic.)

Although the specific designs shown by Thiokol were not directly applicable to the TRW Systems study because of payload weight variations, parametric data presented enabled a comparison to be made of an optimum Thiokol type motor with the modified Minuteman Wing VI motor for the 1971-73 mission. It should be noted that the Thiokol motor characteristics closely approximate the JPL solid propellant rocket system design data* and, therefore, should not necessitate additional comment. On the other hand, the Aerojet propellant for the Minuteman application would be considered to slightly exceed the specific impulse design guidelines for an aluminized propellant. However, the ANB-3066 propellant is currently in production for the Minuteman and Alcor 1A motors and has thoroughly demonstrated physical and performance characteristics, with the exception of proven long-term space storage capability.

1.1.4 Selection of Solid Motor

From the previous discussion, it was concluded that no existing "off-the-shelf" motors are directly applicable to the Voyager mission. The selection, then, would be between a modified Minuteman Stage II Wing VI motor and a completely new development.

* "Design Data for Candidate Voyager Spacecraft Propulsion Systems," JPL, 12 November 1965.

Results of the comparison indicated that, for the same weight allocation, a custom Thiokol type motor would provide within 1 per cent of the total impulse that could be achieved with a modified Wing VI motor. This is due to the fact that the lower mass fraction of the modified Wing VI motor is substantially compensated by the higher performance characteristics of the Aerojet propellant. From the standpoint of reliability and cost, historical data available to TRW Systems indicates that Thiokol's position may reflect an overly optimistic view.

Similar tradeoffs exist with respect to the Aerojet custom solid configuration and, further, Aerojet has indicated that higher costs and a longer development schedule would be involved with a custom motor program.

It is recognized that a custom motor has many advantages for the Voyager mission applications, and that the degree of modifications required to adapt the Minuteman Wing VI motor is considerable. However, based on all information available, it was decided that the modified Minuteman Wing VI Second Stage should be selected as the representative of the solid-engine class of propulsion for further spacecraft integration and design studies for the 1971-73 and 1975-77 Voyager mission application.

1.2 Midcourse Propulsion and Orbit Trim Alternates

Either a liquid monopropellant or a liquid bipropellant system could be used to meet midcourse and orbit trim propulsion requirements. However, a qualitative evaluation of these alternate midcourse propulsion system approaches with respect to the JPL Competing Characteristics criteria reveals an obvious choice. From the standpoint of overall propulsion reliability and development cost, the monopropellant system offers a significant advantage. It is inherently less complex and is well within the state of the art for space application, as recently demonstrated during the successful Mariner 4 mission. On the other hand, the development of a bipropellant system for long term space application requires significant development. It may be argued, of course, that a complete development program would not be needed since engines and components are currently available or under development which could be adapted to the Voyager mission requirements. One such bipropellant engine is the C-1, under development by the Reaction Motors Division of Thiokol, which is intended for universal space applications. Historical experience would indicate that the use of qualified components will significantly reduce the magnitude of the total development program for a bipropellant system, although the required evaluation and verification of the complete system and subsystem interactions including feed system, controls, etc., remains an appreciable program. A thorough and comprehensive qualification test program, to the operational duty cycle of the particular mission application, is mandatory, of course, whether components are to be developed or are currently available.

In addition to the cost and schedule disadvantages of the bipropellant, it was determined that (1) the performance improvement it offered was not enough to meet any stated requirement for 1971-73 not met by the monopropellant system, and (2) the reliability assessment of the bipropellant system results in a probability of successful operation inferior to that of the monopropellant. The performance comparison is included in the results of Table 3. The solid plus monopropellant system meets the required orbit insertion ΔV of 2.00 km/sec in 1971-73. While the solid plus bipropellant produces 2.10 km/sec, this is still short of the "desired" value of 2.20 km/sec. Furthermore the use of a bipropellant system enables a solid sized sized for 1975-77 to provide only 1.91 km/sec for the 1971-73 missions, so, whether monopropellant or bipropellant is used, the dual solid motor development of JPL alternate (a) is necessary.

The reliability assessments of Appendix A show probabilities of success of .949 and .935 for the solid plus monopropellant and bipropellant, respectively. The difference is attributable principally to the additional components in the propellant feed system of the latter, and to the necessity of bellows rather than bladders for use with N_2O_4 for positive expulsion. It is not surprising that this difference exists, nor that the solid plus bipropellant combination is inferior to the alternate based on the LEM descent propulsion stage. For it is inherently illogical to consider the application of a solid motor concept for the retro-maneuver and a liquid bipropellant for the midcourse propulsion system, when a single bipropellant liquid system of comparable complexity could accomplish all propulsion functions for the Voyager mission. With this philosophy, further consideration of a liquid bipropellant system for midcourse and orbit trim propulsion was discontinued and the design study effort was devoted to evaluation of the monopropellant with the solid motor concept.

1.2.1 Midcourse Propulsion System Description

A midcourse propulsion system similar to that selected in the Task A study was chosen for the combination propulsion concept. Figure 10 shows a schematic of the system, which differs primarily from the Task A midcourse propulsion system by the provision for a regulated pressure feed system rather than a blowdown mode of tank pressurization. In Task A it was estimated that a weight penalty of 18 pounds would be incurred by the utilization of blowdown pressurization. This was considered to be justified in view of the inherent simplicity and potentially higher reliability of this approach. However, the weight penalty associated with the blowdown mode for the current study is considerably higher (160-250 pounds). Also, from the standpoint of tank size, the regulated pressure system offers a definite advantage with respect to spacecraft installation. Although the regulated system will be more complex than a blowdown system and, hence, subject to more failure modes, by the application of redundancy and proven components the system will meet mission reliability requirements. Another major difference with respect to the Task A design approach is in the use of four engines rather than a single engine due to spacecraft installation and thrust vector control considerations. The engines are located symmetrically around the solid motor nozzle exhaust and are fired in opposing pairs, with one pair providing a redundant mode of operation.

The thrust chamber design is similar to that shown in Figure 7-18 of Volume 5 for the Phase 1A, Task A study report, except that a thrust level of 100 pounds has been selected. Total burn times for the midcourse and orbit trim maneuvers will be nominally 1000 and 218 seconds, respectively, at this thrust level. A spontaneous catalyst, Shell 405, is provided for engine start, thus precluding the complication associated with

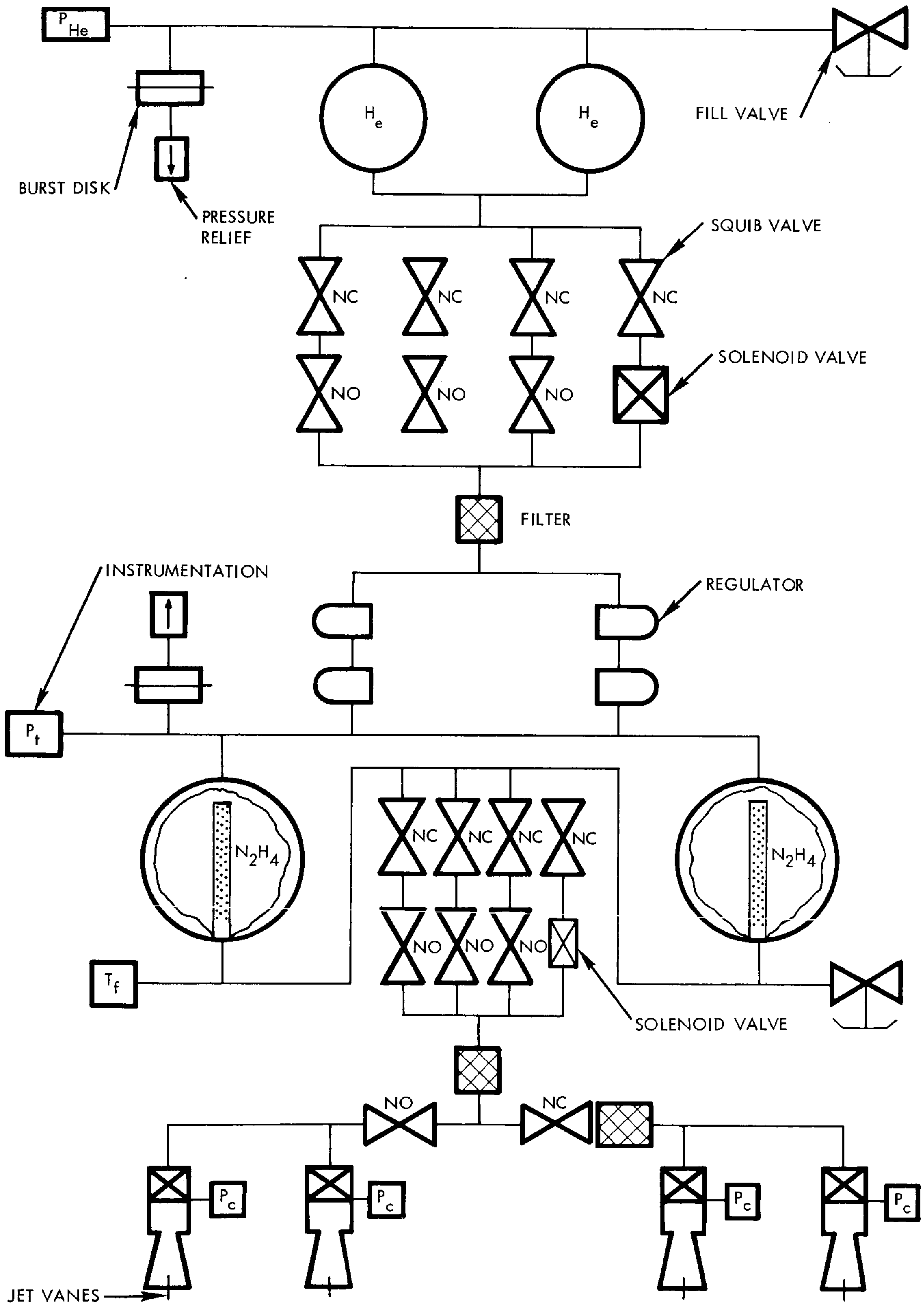


Figure 10. Schematic Diagram of Midcourse and Orbit Trim Propulsion System

N_2O_4 start cartridges. Although the suitability of this catalyst for long space vacuum storage periods has not been demonstrated, currently there is no evidence to indicate problems for an application such as Voyager.

Thrust vector control is provided by means of motor-driven jet vanes located at each nozzle exhaust, which deflect the jet as required for pitch, yaw, or roll. Jet vane actuators of this type are available as proven hardware and would be expected to provide satisfactory vehicle control characteristics.

The 43.6-inch-diameter propellant tanks are fabricated of 6AL-4V titanium alloy and are pressurized to a nominal value of 250 psia, thus assuring satisfactory thrust chamber performance. With the 22.8-inch-diameter helium bottles pressurized initially to 3000 psia and the system loaded with 2081 pounds of N_2H_4, the inert weight of the midcourse and orbit trim system will be 418 pounds and the total weight will be 2500 pounds. The total weight for 1975-77 will be 3730 pounds.

Other design features of the midcourse propulsion system include:

- Squib valves to enable three separate burn operations, and a normally-closed squib valve and solenoid valve in series to enable additional orbit trim maneuvers, if desired
- A quad redundant regulator arrangement for maximum feed system reliability
- A collapsible bladder and perforated standpipe for positive expulsion
- Appropriate pressure relief valves, fill, drain, and vent valves, filters, and flow control valves to assume high reliability, safety, and operational characteristics in compliance with mission requirements
- All system joints will be either welded or brazed, to ensure leakage resistant characteristics compatible with extended space storage.

1.2.2 Midcourse Propulsion System Performance and Operational Characteristics

A total impulse of 489,000 lb-sec is provided in 71-73 to enable two interplanetary velocity corrections totaling 200 meters/sec and one orbit trim maneuver of 100 meters/sec. Based on the requirement for performing a minimum ΔV correction of 1 meter/sec during interplanetary transit, the minimum impulse bit capability of the system must be less than 1636 lb-sec at the end of the second interplanetary velocity correction for the configuration with the flight capsule removed. Four 100-pound thrust engines would require a square wave burn time of 4.08 seconds to achieve this minimum impulse bit, which is well within the state-of-the-art capability for monopropellant systems of this size. Similarly, the requirement for a ΔV accuracy of 0.04 meter/sec can be easily met with engines of this thrust level.

The approach for selection of the system configuration was based on results of tradeoff studies conducted during Task A, modified as required to achieve high reliability and the functional capability for a regulated system with four engines. All electrical power and sequencing functions will be integrated with the spacecraft CS and C system. As indicated by Figure 10, redundancy and operational flexibility are provided by the arrangement of components. Squib valves provide for a nominal of three firings and assure minimum leakage during coast. These valves are supplemented by solenoid valves which also permit improved engine start and shutdown characteristics.

1.2.3 Midcourse Propulsion System Problem Areas

Several potential problem areas exist which would need further investigation and resolution during the development of the system. These include:

- The effects of prolonged exposure of the Shell 405 spontaneous catalyst to a space environment will require further investigation
- The relatively large fuel tanks require an extensive tank and expulsion bladder development program
- Potential leakage problems associated with the storage of high pressure helium for prolonged periods.

However, none of these problems would be considered seriously detrimental to the application of a monopropellant hydrazine midcourse propulsion system.

1.3 Summary

From the foregoing, the preferred combination solid orbit insertion and liquid midcourse subsystem would consist of a modified Minuteman Wing VI Stage II motor and monopropellant hydrazine system. This overall propulsion arrangement would have the following characteristics relative to criteria enumerated in 2.1 through 2.4 of Section III:

- Probability of Success. The reliability assessment of Appendix A shows a probability of successful operation of the solid motor system (including its thrust vector control by liquid injection) of .9743. The corresponding result for the monopropellant system for midcourse and orbit trim maneuvers is .9946. These are relatively high reliabilities for primary and auxiliary (vernier) propulsion systems.

- Performance. The combination system should meet performance requirements as stated for the 1971-73 mission. Performance for 1975-77 is considered marginal, even if the solid motor for 75-77 employs a Beryllium-loaded propellant.

- Cost. Although development and ultimate costs for the combination system would not be a major portion of the entire Voyager program, the cost is the highest of the specified alternates. Appendix B indicates the cost of propulsion system development to be $36.2 million, and propulsion system production for the 1971 mission $18.8 million.

- Flexibility. The combination system has significantly less flexible characteristics than a single bipropellant system for accomplishment of all propulsion operations. Any change to the solid motor total impulse requirement would seriously affect the motor design and could require a new motor development. Further, variations of total orbit insertion impulse and/or thrust level to accommodate modified mission objectives during interplanetary transit are not available as would be the case for a bipropellant system.

Direct comparisons of these characteristics with the other options are given in Section VI.

2. LEM DESCENT PROPULSION STAGE

The inadequacies of the present LEM descent propulsion stage can be readily remedied by minor modifications and additions to the system. These do not require extensive redevelopment or requalification of the existing hardware. This section describes the basic LEM descent stage, identifies basic problem areas and inadequacies, presents a design solution for overcoming each of the cited problem areas, and describes the modified LEM descent stage which was eventually selected as the best choice for the Voyager mission.

2.1 LEM Descent Propulsion Stage Description

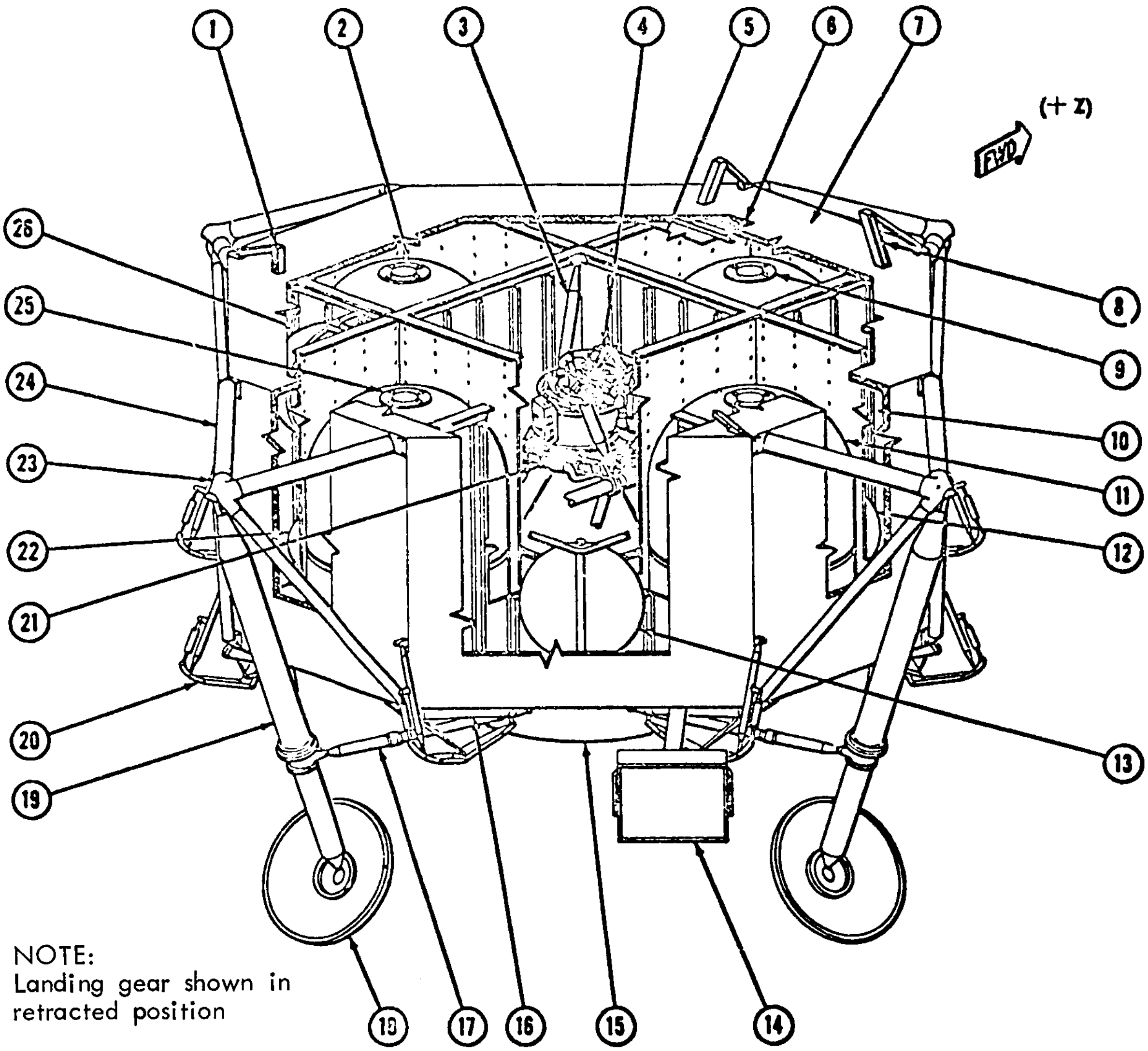
The Lunar Excursion Module Descent Propulsion Stage (LEMDS) is a part of the Apollo system. This system is designed to place three men in lunar orbit, land two men on the lunar surface, and return three men to earth. The total mission duration is approximately 8 days.

The primary function of the LEMDS is to deorbit and soft-land the two men and the ascent stage on the lunar surface. A pressure-fed propulsion system with a variable thrust rocket engine is being developed for the descent stage. The system uses N_2O_4 and 50/50 N_2H_4 and UDMH for propellants. Initial flight tests of the LEMDS are scheduled for 1969; the first lunar landing is scheduled for 1970.

The total weight of the LEMDS is approximately 23,000 pounds, of which 18,000 pounds are propellant. The 5000 pounds of inert weight includes items such as structure, crew environmental and life support equipment, electrical equipment, landing equipment and electronics, and the propulsion system. A general arrangement of the existing LEMDS is shown in Figure 11; the over-all stage dimensions and descent engine installation technique is presented in Figure 12.

The propulsion system will utilize either an ambient or a supercritical helium pressurant storage system. Both systems are undergoing concurrent development efforts. The four titanium propellant tanks, two oxidizer and two fuel, are pressurized from a common pressure regulation assembly. Two explosive valves are used to isolate the propellant/pressurization system during storage. The entire propellant/pressurant feed system is of welded or brazed construction, except for the propellant tank outlets.

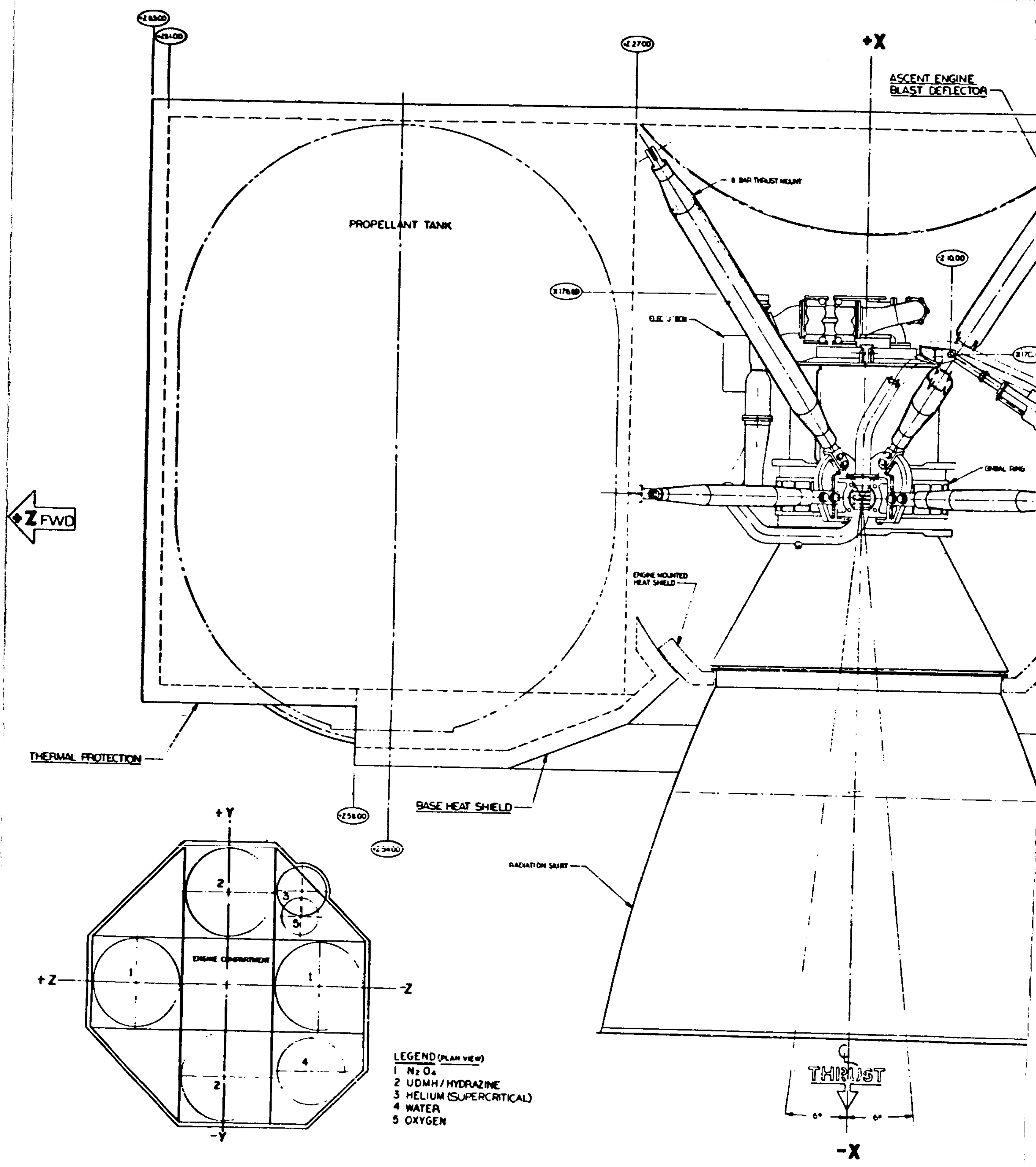
Cn-off propellant flow to the variable thrust rocket engine is controlled by series-parallel valving. These valves are mechanically-linked ball valves, utilizing a common pilot actuated hydraulic (fuel) actuator.



NOTE:
Landing gear shown in retracted position

1. Aft interstage fitting
2. Fuel tank
3. Engine mount
4. Decent engine
5. Structural skin
6. Insulation
7. Thermal shield
8. Forward interstage fitting
9. Oxidizer tank
10. Scientific equipment bay
11. Fuel tank
12. Water tank
13. Helium tank
14. Landing radar antenna
15. Descent engine skirt
16. Truss assembly
17. Secondary strut
18. Pad
19. Primary strut
20. Lock assembly
21. Gimbal ring
22. Oxygen tank
23. Adapter attachment point
24. Outrigger
25. Oxidizer tank
26. Hydrogen tank

Figure 11. LEM Descent Stage , General Arrangement


+X
ASCENT ENGINE
BLAST DEFLECTOR
PROPELLANT TANK
GIMBAL RING
ENGINE MOUNTED
HEAT SHIELD
+Z FWD
THERMAL PROTECTION
BASE HEAT SHIELD
RADIATION SKIRT
THRUST
6°
6°
-X
+Y
-Y
+Z
-Z
LEGEND (PLAN VIEW)
1 N_2O_4
2 UDMH / HYDRAZINE
3 HELIUM (SUPERCRITICAL)
4 WATER
5 OXYGEN

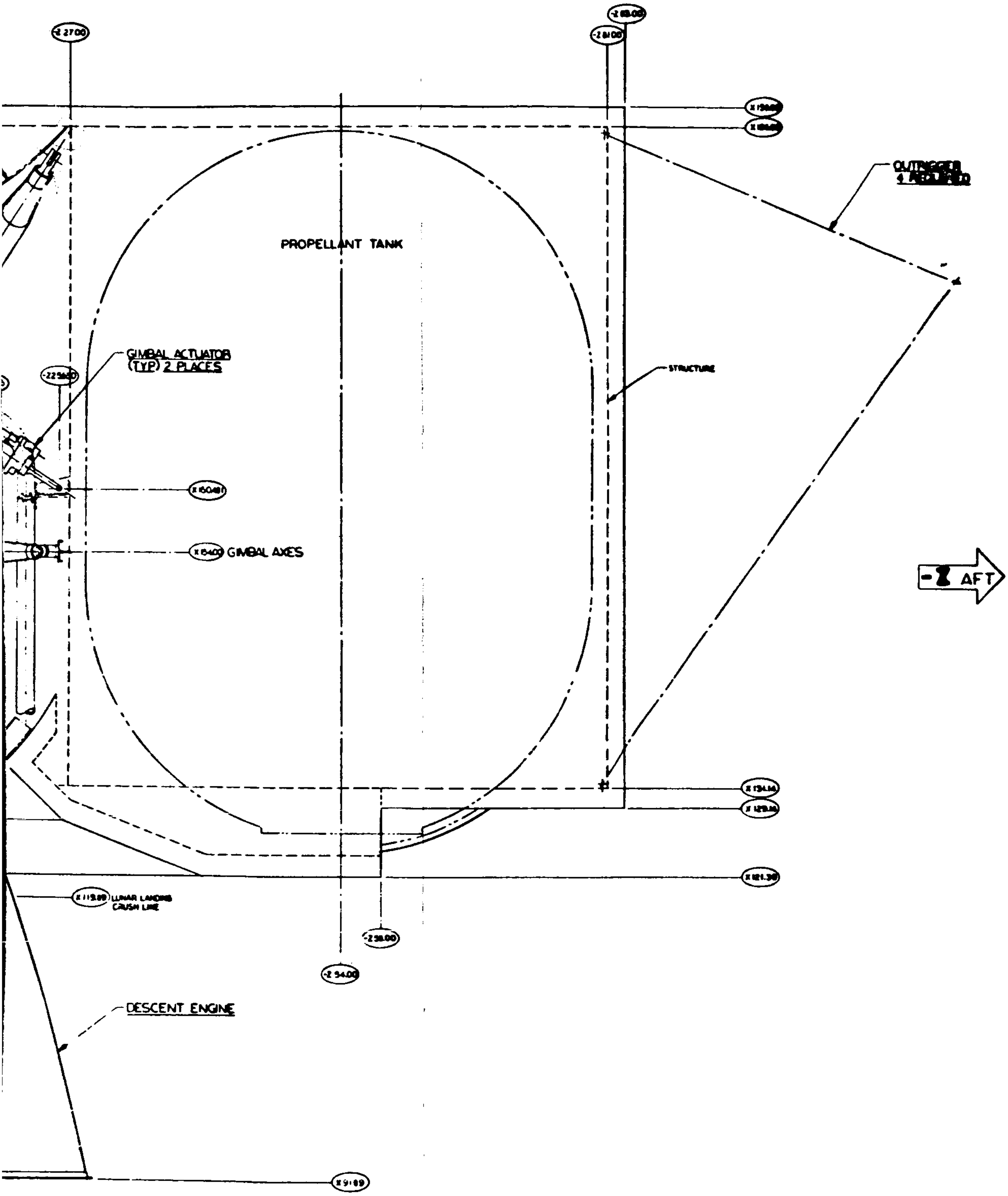


Figure 12. Descent Propulsion Engine Installation

2

Propellant flow rate during throttled operation is controlled by variable area, cavitating venturis. A variable area concentric injector is used to maintain satisfactory injector hydraulic characteristics over the 10:1 throttling range. The variable area injector and cavitating venturis are mechanically linked and utilize a common electromechanical actuator for positioning.

The single ablative thrust chamber assembly with a radiation cooled nozzle extension has an operational life in excess of 1000 seconds. At 105 psia chamber pressure and an oxidizer-to-fuel mixture ratio of 1.6, the engine develops 10,500 lbf of thrust at an I_{sp} of approximately 305 seconds. Thrust vector control is provided by gimballing the rocket engine about the throat plane using electromechanical gimbal actuators.

The over-all configuration of the LEMDS was dictated by the Apollo system requirements of accomplishing a manned lunar landing and return to earth. The propulsion subsystem reliability is estimated to be in excess of 0.9988 for the Apollo mission. This reliability level is achieved through redundancy of components and conservative design concepts.

LEMDS subsystems and components were also dictated by the Apollo mission. A major portion of the crew environmental control and life support water, hydrogen, and oxygen, are located in the descent stage. Landing gear and portions of the landing radar and electronics are also contained in the descent stage, as are other specialized electronics, and scientific equipment designed especially for the Apollo mission. This equipment is not necessary for a Voyager mission. Propellant settling and spacecraft attitude control are provided by an attitude control system located in the LEM ascent stage, which is mounted above the LEMDS.

2.2 Applicability of Unmodified LEMDS to Voyager

The LEMDS is designed for a Saturn C-5 launch, thereby simplifying spacecraft structural development. Ample propellant capacity is available in the existing propellant tanks. The variable thrust rocket engine assembly provides a 10:1 range of thrust levels in addition to an operational life of over 1000 seconds.

Fundamental differences, however, between the Voyager and Apollo spacecraft designs and mission requirements, primarily associated with long life, multiple starts, and booster system interactions preclude the use of an unmodified LEM descent stage for Voyager. In addition, the LEM descent stage includes subsystems, such as environmental control, life support, and communications, which are not required for Voyager. However, the modifications required to adapt the LEM descent stage are relatively low cost and low risk development tasks.

2.3 Problem Areas

When considering the application of the LEMDS to Voyager, specific problem areas became apparent. These problem areas, as discussed in the following paragraphs, are:

- Excessive Stage Weight

  The 5000-pound LEMDS contains several subsystems that are applicable in whole or in part only to the Apollo mission. Existing subsystems such as pyrotechnics, stabilization and control, navigation and radar, crew provisions and environmental control, landing gear, and electronics must be modified and/or removed. In addition, certain rocket engine subsystems, such as those required to provide continuous rocket engine throttling, are not necessary and may either be removed or replaced by lightweight, less complex units.

- Long Term Space Storability (approximately 1 year)

  The present LEMDS requirements specify approximately 45 days operational life in the translunar environment. Although this storage life is not sufficient for the 7- to 9-month Voyager mission, system modifications in the areas of thermal and meteoroid protection to increase storage life in space are essentially straight-forward, "beefing up" type of modifications. Stress corrosion in the titanium alloy nitrogen tetroxide tank is currently a critical problem. This phenomenon is the subject of intensive investigation, and techniques such as propellant additives, tank coatings, and tank material substitution, are being investigated. Whereas demonstration of design adequacy will require a long and fairly costly program, the risk involved in the problem is believed to be low.

  Propulsion system gas and propellant leakage problems should be minimal. The LEMDE feed system uses either brazed or welded construction with few exceptions. The ball type propellant valves which represent the major leakage points in the system will be replaced by explosive valves for the Voyager mission.

Propellant diffusion into the pressurization system may be a storage problem; however, the magnitude of this problem during an extended zero-g exposure has not been accurately established. The zero-g environment could result in the pressurization system lines and check valves being exposed to propellant liquids as well as vapors for a period of 7 to 9 months. Under these conditions, propellant diffusion through the check valves could contaminate the pressurization system. Therefore, it may be necessary to positively isolate the propellant from the pressurization system during extended periods of non use.

- Rocket Engine Restart Capability

  The existing LEMDE is dependent on the ascent stage reaction control system to provide propellant settling for zero-g engine starting. Since the Voyager mission requires engine restart, it will be necessary to add this capability to the LEMDS.

- Thrust Vector Control

  The existing gimbal system is designed only to maintain thrust vector/vehicle center of gravity alignment with the Ascent Stage mounted above the Descent Stage. The ascent stage attitude control system orients the entire LEM spacecraft to keep the descent engine thrust vector aimed in the appropriate direction. This control technique presents two problems in applying the LEMDS to Voyager. They are: (1) The existing gimbal actuators do not have adequate power or response to perform the vehicle control functions; therefore, actuators with higher power and faster response are required. (2) For the Voyager orbit trim after the capsule is removed, the center of gravity will descend below the LEM descent stage gimbal axes. This makes it necessary to lower the rocket engine assembly.

- Rocket Engine Radiation Heat Flux

  The heat flux from the radiation-cooled nozzle extension to the solar cell array is a potential problem. Solar cell overheating and subsequent malfunctions could result during the long duration orbit insertion maneuver. Therefore, it will be necessary to shield or reduce the heat flux from the engine nozzle to the solar cell array.

- Cold Welding

  All components of the LEMDE are designed to operate in the vacuum environment and will be qualified for the Apollo mission. However, because of the longer duration of the Voyager mission, it is anticipated that all moving components must be requalified to the Voyager storage requirements and any new components must be protected against cold welding.

2.4 Recommended Modifications and Their Feasibility

The problems identified in the previous paragraphs are solved by modifying the basic LEMDS. Since the performance capability of the stage is adequate, changes directed toward performance improvement and involving additional development efforts or program risks are not considered. Modifications are recommended only where dictated by Voyager mission requirements or if significant reliability improvements could be achieved. In areas where more than one solution was available to correct a problem, the minimum risk approach is selected.

2.4.1 Modifications to Reduce Stage Weights

It is feasible and practical to remove LEMDS subsystems that are not required for the Voyager mission. All life support, landing, and LEM electronic equipment are to be removed. Propulsion subsystems not required for the Voyager mission, i.e., components to provide continuous throttling, are removed and replaced by a lightweight electromechanical actuator for injector pintle positioning. The modifications result in a weight reduction of 1800 pounds, thus lowering the LEMDS propulsion system dry weight to approximately 3200 pounds.

The remaining LEMDS hardware represents propulsion system and structural components only. The structure is required to support the propulsion system components and is also used for mounting spacecraft subsystems such as electronics, environmental control, communications, power supply, etc.

2.4.2 Modifications to Achieve Long Term Space Storability

Modifying the propulsion system operational sequence, propellant valving, and the thermal and environmental protection systems as shown in Volume 2 is recommended to achieve long term space storability. Using a low pressure blowdown mode of operation for midcourse maneuvers reduces tank pressures during the 4- to 8-month interplanetary cruise phase, thus minimizing stress corrosion problems. The variable thrust capability of the LEMDE makes this operational mode feasible. This operational mode has the advantage of allowing positive isolation of propellant from the pressurization system for a major part of the Voyager mission. Removing the existing multicycle propellant on-off valves and incorporating explosive and small solenoid valves significantly reduces the potential propellant leakage problem. This modification is relatively straightforward since qualified components are available. Modifying the existing LEMDS thermal and environmental protection system for the Voyager mission is also feasible.

The blowdown mode of operation offers a significant increase in reliability potential. This mode of operation for the midcourse maneuvers insures that propellant tank pressures remain less than 125 psia for the 4- to 8-month interplanetary cruise. Lower tank pressures reduce the tank stress levels, and low stress levels are desirable from a stress corrosion as well as an over-all feed system reliability standpoint. Low storage pressure also minimizes the potential of propellant leakage. Since pressurization is not required from an external source during blowdown, the propellant remains isolated from the pressurization system during the interplanetary cruise. (The existing LEMDS uses explosive valves to provide isolation during translunar cruise. However, long term storage is not required during the operational phase.)

The existing LEMDS and LEMDE are especially adaptable to the blowdown operational mode. The propellant tank capacity is 18,000 pounds and only approximately 12,000 pounds are tanked for the Voyager mission. Therefore, 33 per cent ullage is available initially. The blowdown maneuvers (midcourse) consume approximately 1400 pounds of propellant resulting in a post-blowdown ullage in the order of 40 per cent. From an initial pressure of 125 psia, the tank pressures decay to approximately 90 psia resulting in a thrust decay of approximately 20 per cent (200 lbf). These pressure levels are adequate to provide stable, high performance with the LEMDE variable area injector positioned at the low thrust setting. Following blowdown operation when the tanks are fully pressurized, the correct injector inlet pressures are provided by orifices in the propellant flow lines.

Low pressure storage of N_2O_4 is the most direct way of minimizing stress corrosion problems with titanium tanks. This may even prove to be an adequate solution, although tank coatings or platings, preliminary passivation or propellant additives to minimize stress corrosion are possible further measures to consider.

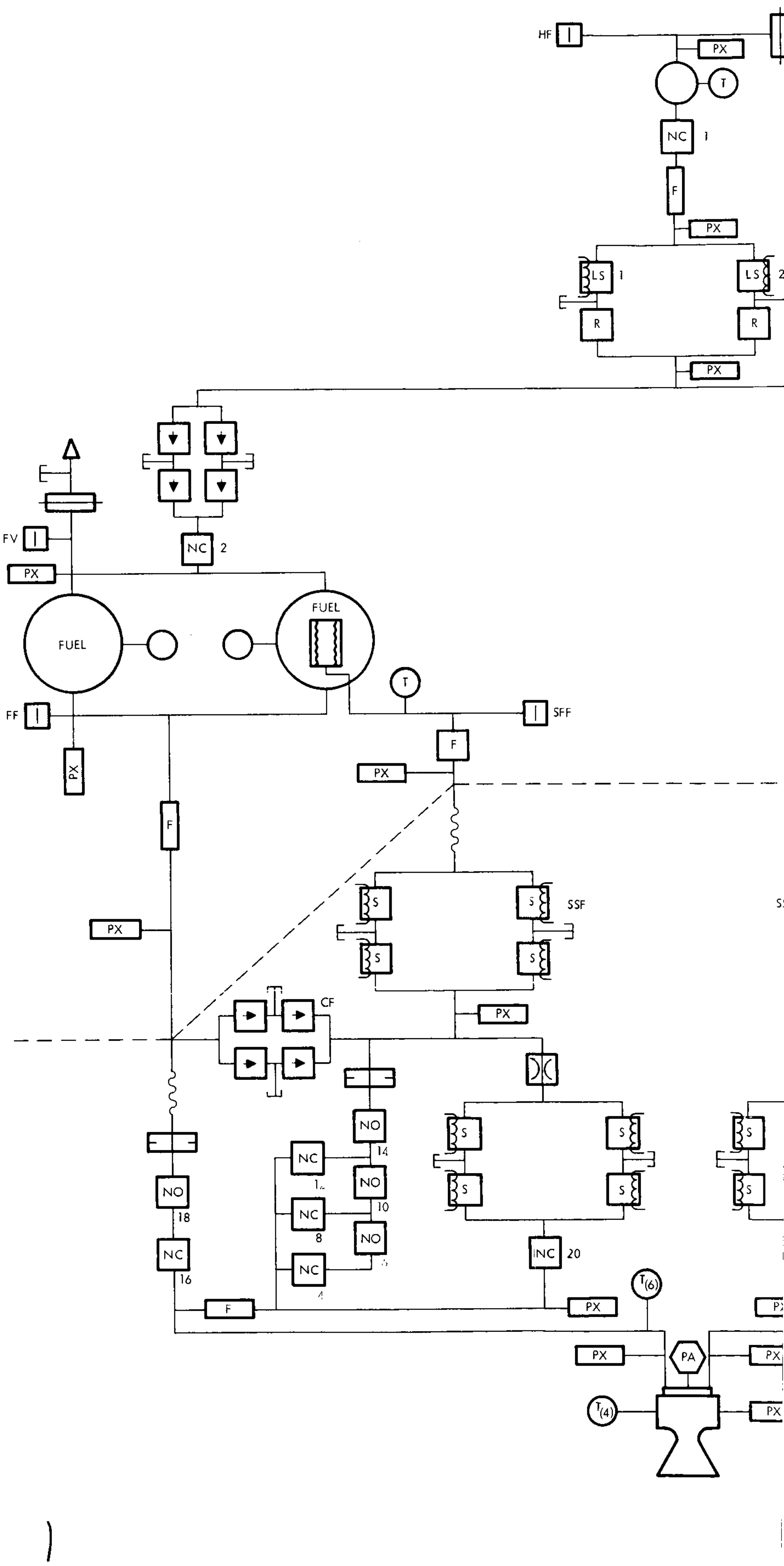
The alternative of resizing the tanks, while saving weight, would complicate the valving and plumbing, reduce mission flexibility (for on-loading), and require more development.

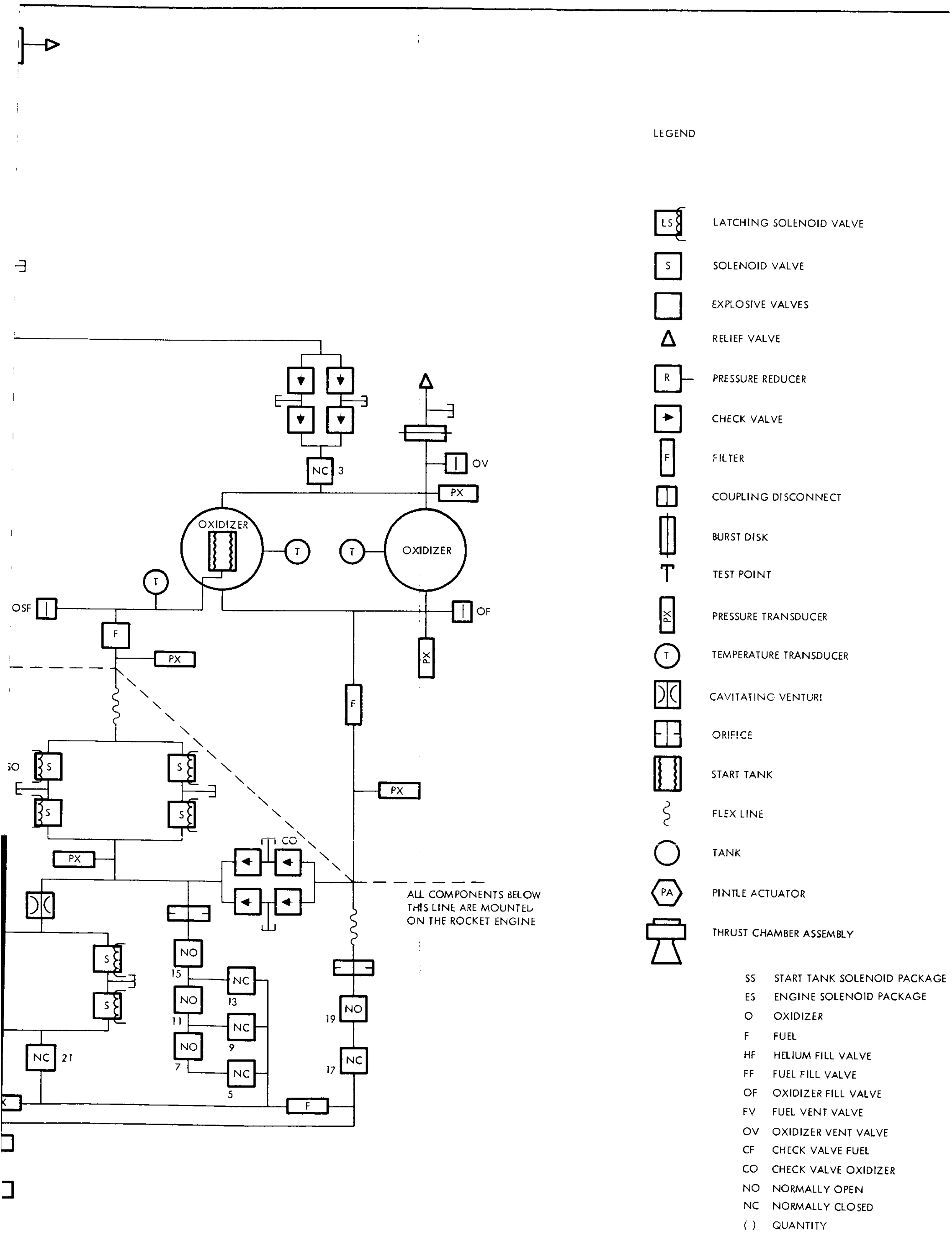
Eliminating the existing LEMDE propellant flow system and using explosive and solenoid actuated valves is recommended to minimize propellant leakage problems. Explosive valves provide positive sealing during the interplanetary cruise. During orbital operations, the one-half inch solenoid valves are recommended to provide additional flexibility. Leakage through these valves during this phase is minimized by the redundant arrangement of the valves. The fast acting explosive and solenoid valves are also suitable for the low, repeatable impulse bits required for the Voyager mission.

The recommended valving arrangement is shown in Figure 13. Twelve single-squib dual-bridgewire explosive valves (4 through 15) provide propellant on-off control for three midcourse maneuvers. Four larger explosive valves (16 through 19) provide orbit insertion propellant on-off flow. Orbital trim maneuver propellant flow is controlled by two series-parallel, quadredundant solenoid valve packages (ESF and ESO). These solenoid valve packages are isolated during the interplanetary cruise by two explosive valves (20 and 21).

Start tank propellant flow is also turned on and off by quad solenoid valves (SSF and SSO). Two series-parallel check valve packages (CO and CF) are located between the engine main tank propellant inlet lines and the start tank solenoid valve outlets. These check valves have a sufficiently high cracking pressure to prevent main tank flow into the engine start plumbing until the main propellants are settled and the start tank flows are terminated.

Explosive valves are selected over multicycle valves because of their positive sealing characteristics, low power drain, response, demonstrated reliability, and off-the-shelf technology. Positive sealing and


HF
PX
T
NC 1
F
PX
LS 1
LS 2
R
R
PX
NC 2
FV
PX
FUEL
FUEL
T
SFF
FF
F
PX
PX
F
PX
SSF
S
S
S
S
CF
PX
NO
14
NO
10
NO
NC
NC
8
NC
NO
18
NC
16
F
S
S
S
S
NC 20
T(6)
PX
PX
PA
PX
T(4)
PX



2

Figure 13. Voyager Propulsion Subsystem Plumbing Schematic Diagram

reliability are considered as the most important advantage for the Voyager mission. Multicycle valves are prone to leakage which may result from seat damage due to contaminants in the propellants. Also, the multicycle valves have more points of potential malfunction. For these reasons the explosive valve is selected as the minimum risk approach toward achieving reliable propellant storage during the interplanetary cruise.

Solenoid valves are used to provide all engine starts after the mid-course maneuvers. The multicycle feature of these valves provides increased mission flexibility and allows alternate modes of operation in event of malfunction of explosive valve firing circuitry or explosive valves. These valves are arranged in a series-parallel configuration to increase reliability. They are used for orbit insertion and orbit trim maneuvers, spanning a comparatively short time interval, so that the mission is not jeopardized by potential cumulative leakage.

2.4.3 Modifications to Achieve Rocket Engine Restart

Three possible methods for achieving propulsion system restart were considered: (1) use of ullage orientation rocket motors fed from auxiliary positive displacement propellant tanks, (2) use of positive displacement start tanks located within the propellant tanks, and (3) use of separate positive displacement start tanks.

Installing positive expulsion tanks employing metal bellows within one of the two main fuel tanks and one of the two main oxidizer tanks was selected. This is the minimum weight approach and high reliability is retained since the internal mounting eliminates the requirements for separate start tank pressurization plumbing and high pressure (250 psia) tanks. Also, increased mission flexibility is achieved by sizing the start tanks to provide more than the eight required restarts. This restart technique is compatible with both the blowdown and pressurized operational modes due to the valving arrangement described earlier.

The start tanks are thin wall cylinders with metal bellows. The bottom of each start tank cylinder is sealed to the respective main tank aft closure and open at the top. The open top allows the pressure in the main tank to act directly on the metal bellows. Thin wall start tank construction is satisfactory since the start tanks are located inside the main tanks, thus reducing the pressure differential across the start tank wall to a value of less than 10 psi.

The internal start tanks are located on the main tank aft closures and are installed in a similar technique to that used for installing the main tank baffles. Although the tank internal configuration is essentially unaffected, it is necessary to provide an additional outlet port on the main tank closure for the start tanks. However, this modification is straightforward. The balance of the start tank plumbing is described in the previous section (see Figure 13).

Other techniques to achieve restart require extensive development efforts, thus increasing over-all program costs and risks. A separate propellant settling reaction control system adds increased cost and system complexity. Even if the existing ascent stage attitude control system were adapted to the Voyager mission, the over-all system reliability would be degraded due to the additional number of components, i. e., valves, regulators, engines, etc.

It is also possible to locate the start tanks externally, in the space provided by lowering the rocket engine. This approach, however, results in a heavier system due to the high pressure (250 psia) tanks. The high pressure storage degrades system reliability, since both the internal and external tanks would employ the same bellows expulsion technique to take advantage of developed technology. The external tanks would require a pressure supply system which would also tend to degrade system reliability when compared to the internal configuration.

2.4.4 Provision for Two-Level Thrust Operation

With the removal of the actuating control mechanisms for continuous throttling capability, a lightweight actuator is introduced for two-position injector pintle positioning. This saves weight and increases reliability without sacrificing operational flexibility for Voyager requirements. The two thrust levels selected are 1050 and 7750 pounds. The lower level, the lowest achievable by the LEM descent engine, is chosen to provide the smallest impulse bit and the smallest nonproportional shutoff error to meet the needs of the Voyager midcourse and orbit trim maneuvers. The higher level, less than the 10,500-pound maximum of the engine, is chosen to limit the maximum deceleration during orbit insertion to 1 g, to simplify appendage mechanical design, while maintaining high performance of the engine. (The I_{sp} at 7750 pound thrust is no less than at 10,500 pounds.) Total operating time for Voyager, using these two thrust levels, is below the engine design lifetime of 1200 seconds.

2.4.5 Modifications to Reduce Rocket Engine Radiative Heat Flux

Replacing the existing LEMDE radiation cooled nozzle extension with an ablative extension is recommended to reduce the radiated heat flux. The ablative nozzle does not require any moving parts as would deployable radiation shields. This approach also provides the capability of simply extending the ablative portion of the existing nozzle divergent section, heat flux levels permitting, and using a section of existing radiation nozzle to reduce system weight.

The ablative extension is attached at the same point and has the same internal contour as the existing radiation extension. Refrasil phenolic ablation material and a fiberglass overwrap will be used. Standard fabrication techniques developed on other engines of the LEMDE thrust class, e.g., the LEM ascent engine, are satisfactory for this modification.

Other techniques available for reducing the heat flux represent higher risk approaches. The solar cell array must be protected from high heat loads, yet exposed to solar radiation. This necessitates some form of moveable shield; the moving shield requires actuators and would have sliding surfaces. Reliability is degraded by adding multicycle actuation systems and protecting external sliding surfaces from cold welding in the vacuum environment. Packaging the shield would also be a problem. Modifying the engine affords the most practical technique to reduce the radiated heat flux.

The all-ablative extension is recommended at this time. A more detailed thermal analysis may show that only a small part of the existing radiation extension must be replaced, or that insulation or stand off shielding to the existing extension is adequate. Until additional data are generated however, the all-ablative extension recommendation represents the conservative approach.

2.4.6 Modifications to Provide Thrust Vector Control

Two modifications are recommended to provide thrust vector control. The low power gimbal actuators are replaced by higher powered units and the rocket engine assembly is lowered approximately 36 inches. The higher powered actuators provide the required gimbal rate and acceleration; lowering the engine prevents the stage center of gravity from passing through the engine gimbal plane.

These modifications are straight forward and do not represent development risks. Actuator technology is well developed and lowering the engine primarily involves engine mount structure and propellant inlet plumbing changes. The existing LEMDS structure is not affected since fortuitously it is already strengthened at the new attach points.

2.4.7 Modification to Ensure Against Cold Welding

The existing LEMDS design and previous modifications eliminated the cold welding problem. The LEMDS is designed for extended vacuum operation in a translunar environment, and modifying the propellant valving and removing the throttle valves eliminated a major portion of the mechanical linkage exposed to vacuum. The electromechanical actuator recommended for positioning the injector pintle is mounted directly on the injector and installed in a sealed metal container.

The only remaining propulsion system moving components that are exposed to vacuum are gimbal components. These items will be flight qualified for an environment of 10^{-14} mm Hg, which is similar to the trans-Martian environment. Therefore, the probability of cold welding problems in the recommended propulsion system is essentially nonexistent.

2.5 PERFORMANCE POTENTIAL OF THE MODIFIED LEMDS

A modified LEMDS has a performance potential in excess of that required for the Voyager mission. The recommended Voyager system has a ΔV capability in excess of 3400 meters/sec if fully loaded. Minimum Voyager propulsion requirements of 2300 meters/sec are obtained by off-loading propellants.

The performance capability estimate is based on using the existing LEMDE injector and thrust chamber in a stepped thrust operational mode. The present LEMDE is capable of continuous throttling over a 10:1 thrust range and, as a result, is readily adaptable to steady-state operation at selected thrust levels within the 10:1 range. For the Voyager mission, the injector will be positioned at a nominal 1050-pound thrust setting for the midcourse and orbit trim maneuvers, and at 7750 pounds for the orbit insertion maneuver. The average delivered I_{sp} for this operational mode is greater than 302 seconds, with 9654 pounds of propellant consumed during high thrust operation (I_{sp} = 305 sec), and 1720 pounds consumed during the low thrust operation (I_{sp} = 290 sec).

Figure 14 presents the ΔV capability of the modified LEMDS as a function of total system launch weight (excluding the planetary vehicle adapter) based on an average delivered I_{sp} of 303 seconds obtainable from the usable propellants. The additional ΔV capability, in excess of that

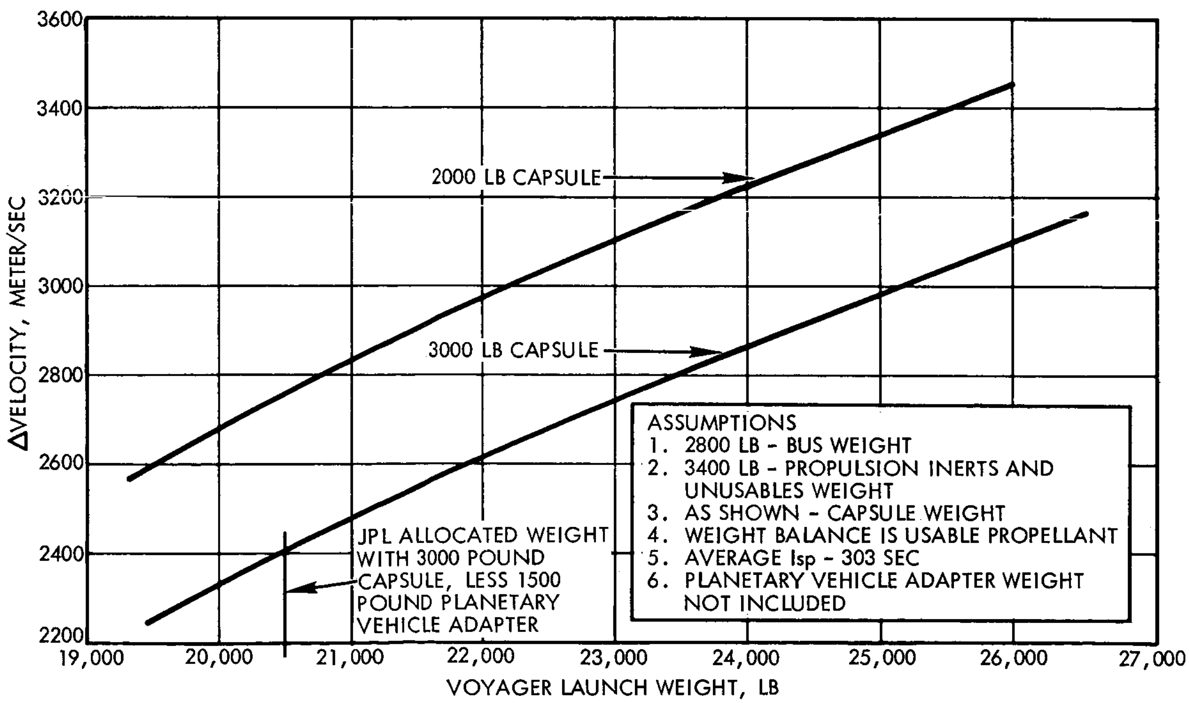


Figure 14. Voyager Performance Capability as a Function of Launch Weight

required for the Voyager mission, provides a significant degree of mission flexibility. This increased mission capability is due to the oversized propellant tanks available on the LEMDS and the extended burn duration capability of the LEMDE.

The minimum impulse bit capability of the modified propulsion system is essentially dependent on the manifold volumes, vehicle control system sensing and command equipment, and valve response times. (Approximately 1.4 pounds of propellant are trapped below the propellant valves. The explosive and solenoid valves allow only a negligible amount of propellant, 0.03 to 0.07 pound, to flow into the engine after thrust termination commands.) Although the LEMDE minimum impulse bit has not been demonstrated, a minimum impulse bit well under the 950 pound-seconds, required for 1 meter/sec ΔV manuevers after orbit insertion, should be easily attainable. The fast response valving also provides an estimated 75 pound-second (3σ) impulse repeatability.

The effect of this 75 pound-second nonproportional error on the accuracy of velocity corrections is shown in Figure 15 as a function of vehicle weight.

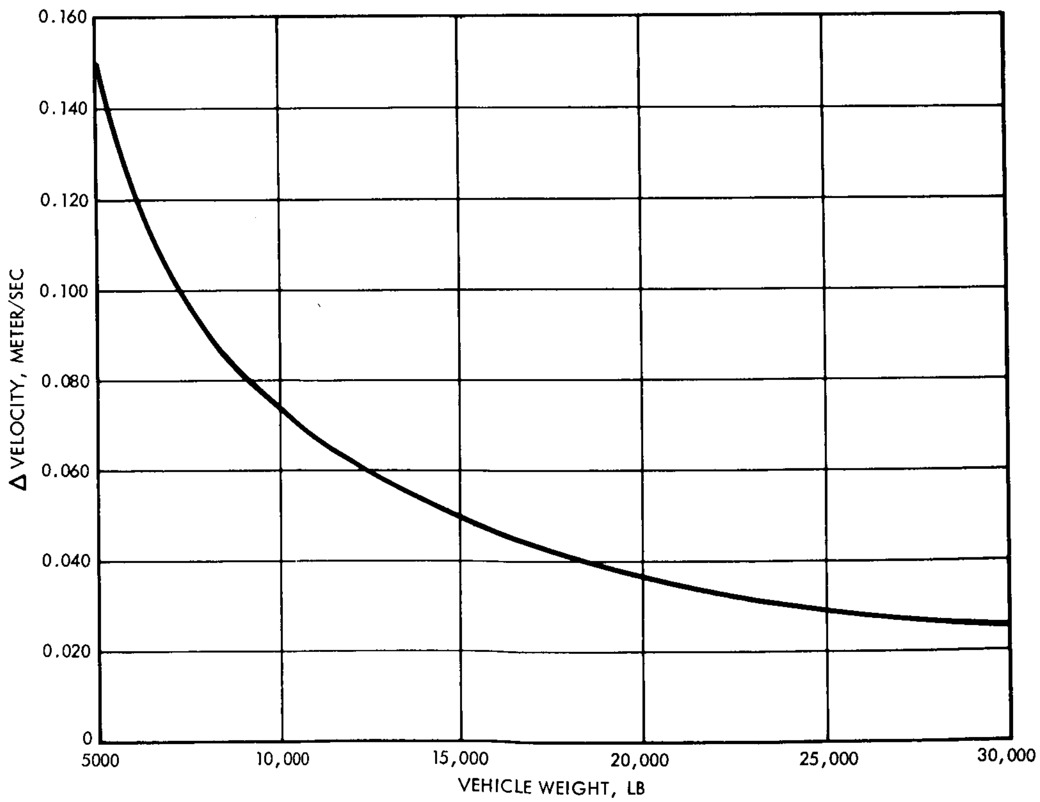


Figure 15. Velocity Increment Nonproportional Error of Modified LEMDS with 75 lb-sec Impulse Bit Repeatability

2.6 Summary

Although the existing LEMDS must be modified to accomplish the Voyager mission, the more expensive LEMDS subsystems and components are satisfactory for use on the modified stage. The modifications that are required primarily involve minor system components and represent little development risk. A major portion of the cost of developing a Voyager propulsion system from the basic LEMDS will be associated with verification testing and requalification efforts.

Reliability is an important factor in configuring a Voyager spacecraft. The over-all reliability potential of the modified LEMDS is adequate for the Voyager mission. High reliability, currently estimated at 0.9988 for the Apollo mission, is achieved by component redundancy and design simplicity: these features are emphasized in the recommended modifications. Also, each recommended modification was selected with reliability and minimum development risk as the primary selection criteria. Appendix A presents two assessments of the LEM descent propulsion stage as modified for Voyager. One, using Grumman failure rates and analytical methods as applied to the Apollo mission, gives a probability of success (for the propulsion subsystem, including engine gimbals, in the Voyager mission) of 0.9913. A second, based on FARADA and TRW failure rates and TRW analytical methods, gives 0.9678. The latter figure may be more meaningful compared with the probabilities of success of the other alternate systems, and is, we belive, more realistic.

The minimum risk philosophy is especially important in the Voyager program since the available launch periods are fixed. The minimum risk philosophy and extensive use of existing hardware significantly reduce the possibility of development program schedule slippages or funding overruns. In all cases, the modifications can be implemented with existing hardware or technology. System development efforts will primarily involve component integration, verification, and qualification efforts.

A Voyager mission also imposes severe storage requirements on spacecraft design. The existing LEMDS is inherently suited to long term space storage. With few exceptions, all pressurant and feed system joints are of brazed or welded construction. The modifications recommended improve the basic storability of the LEMDS by providing positive sealing propellant valving and low pressure propellant storage during the interplanetary cruise. The differences between the trans-lunar and trans-Martian environments require only minor stage modifications to accomodate the 4- to 8-month Earth-to-Mars transit time plus a nominal 6-month period of orbital operations.

The recommended system also maintains the LEMDS design philosophy of redundant operational capability in event of a component malfunction. With the exception of the explosive valves (which have dual bridgewires) and the injector pintle assembly, any one moving part could malfunction without degrading the mission. Even an explosive valve failure can be circumvented by the presence of the solenoid valves normally used only during orbit trim. If the midcourse explosive valves or valve firing circuitry should malfunction and fail to open, the solenoid valves could be used for these maneuvers.

With the recommended modifications, the system is capable of a total velocity increment in excess of that required for the Voyager mission. Minimum velocity capability and accuracy are also within the limits imposed by the Voyager mission. In addition, the oversized propellant tanks and long burn duration capability of the LEMDE allow the basic Voyager spacecraft to accomplish more ambitious future missions. For the Voyager or other missions, the maximum possible flexibility is achieved in the use of propulsive capability because of the single engine, the dual thrust level, and the propellant supply which is common for all propulsion operations.

The costs involved with modifying the LEMDS are considered to be modest in terms of the Voyager program total cost. Recommended modifications utilize state-of-the-art technologies, and the high development cost LEMDS components are used as is. These features not only result in a lower cost program but allow realistic cost estimations to be made. In Appendix B, the costs associated with the use of the LEM descent stage as the Voyager propulsion subsystem include \$20.0 million for development and \$16.9 million for production of nine units for the 1971 mission. The total propulsion system cost is considerably lower than that of any of the other alternates.

3. TRANSTAGE PROPULSION STAGE

This section describes the current Titan III-C Transtage; identifies several significant problem areas associated with the Transtage; proposes design solutions to adapt the Transtage to Voyager requirements; and defines the modified Transtage configuration which was used in the system comparison. The use of Transtage in an unmodified version or with minimum modifications was ruled out of serious consideration because of a combination of shortcomings which in the aggregate would require extensive design modifications and development to meet minimum Voyager requirements.

3.1 Transtage Propulsion Stage Description

The Transtage is the third stage of the Titan III-C standard space launch vehicle. It is 10 feet in diameter and 15 feet long. Total wet weight is 25,338 pounds, and the propellant tank capacity is 22,874 pounds.

The Transtage is composed of two modules: a propulsion module and a control module. The control module, which weighs 2494 pounds, is located forward of the propulsion module. It contains the inertial guidance system, the reaction control system, power sources, separation and destruct systems, environmental control groups, and telemetry. The control module is approximately 56 inches long. It attaches to the propulsion module at Station 133.6. The control module slips over the main propellant tanks of the propulsion module.

The Transtage attitude control system consists of two clusters of three ablative-cooled chambers, termed the yaw-roll modules, and two single aft pointing nozzles. One nozzle of each of the three-chamber units points aft, providing four 45-pound thrust rockets for propellant settling and pitch and yaw control. The reaction control system propellants are contained in positive-expulsion tanks which use a Teflon bladder to separate the pressurizing gas from the propellants. Two 25-pound thrust chambers of each three-chamber unit are mounted in opposition and fire tangentially. These 25-pound thrust chambers are used in opposite pairs for roll control and in like pairs to augment pitch control during powered flight.

The control module is approximately 56 inches long. It attaches to the propulsion module at station 133.6. The control module slips over the main propellant tanks of the propulsion module.

The two main engines are rated at 8000 pounds. The nominal operating pressure is 100 psia and the nominal mixture ratio is 2.0. Propellants are nitrogen tetroxide and Aerozine 50. The rocket engines are gimballed ±6 degrees to provide control during powered flight. The main engine gimbal is an annular ring mounted around the throat of the engine and attached to the engine and the Transtage thrust mount by flexural pivots. An actuator displaces the engine with respect to the flexural pivot to provide yaw. For pitch, the gimbal ring is displaced relative to the Transtage. Hydraulic fluid is provided to the actuator from an integrated motor-pump-reservoir hydraulic unit. Power is provided to the electric motor from batteries located in the control module.

The propellant is contained in two cylindrical titanium tanks with elliptical forward ends and conical bottoms. Propellant is trapped in the conical bottom by means of a dual check valve in the false bottom of the tank to provide a trap for the propellant for multiple zero-gravity restarts. However, this feature has not been demonstrated in flight, and the attitude control motors are fired to ensure propellant settling prior to main engine ignition.

3.2 Applicability of Unmodified Transtage to Voyager

The significant problem areas, which individually do not pose insurmountable obstacles, but which considered collectively indicate the need for considerable development are: long term storability, minimum impulse bit capability and predictability, structural problems associated with basic stage diameter, and transient control problems of multiple engine systems. These problem areas as related to the use of Transtage for Voyager are discussed below.

3.2.1 Long-Term Storability in Space (~1 Year)

The present Transtage was designed for 6-1/2 hours storage in near earth orbit. However, the problems of 30-day storage have been studied by the Martin Company and certain design changes were recommended. These changes have been tested in a high vacuum orbital simulator to obtain equilibrium temperatures in near earth orbits (but without constant sun orientation). Recommended design changes consisted of basically changing the composition and geometry of the external paint patterns, addition of a shroud to the aft end, and removal of various insulations. On the basis of this effort it was concluded that long-term thermal control can be achieved with relatively minor changes.

An additional factor in long-term storability is the question of stress corrosion of titanium alloy 6AL-4V in nitrogen tetroxide, a problem which has been experienced in recent stress corrosion experiments. Detailed experimental investigations at Martin have resulted in the conclusion that there will be no problem in this regard for periods up to 8 hours, and presumably longer on the Transtage. However, there is considerable evidence from other programs which indicates that this is a very serious problem and considerable development may be required before a reliable solution is demonstrated.

The Transtage propulsion system is designed with many mechanical joints which are potential sources of leakage. In addition, each of the components is a potential leakage point. Tables 4 and 5 give 30-day leakage rates that are allowed by specifications at the present time. Existing test procedures verify that these rates have not been exceeded. It can be seen from the data in the tables that 1-year storage would result in considerable gas and liquid leakage primarily through the mechanical joints in the system.

3.2.2 Minimum Impulse Bit and Accuracy

The present minimum impulse bit is 5000 ±1050 lb-sec of impulse per chamber. This impulse would result in a minimum ΔV correction of approximately 4.8 ± 1.0 meters/sec for the 1971 Voyager vehicle and 3.4 ± 0.7 meters/sec for the 1975 vehicle. Since these values are far too high for midcourse corrections, the use of auxiliary motors are a requirement.

Table 4. Transtage Maximum Helium Leakage Rates, Main Propulsion System (Based on Specification Values)

Description	Quantity	Total 30-Day Leakage (lb)
Mechanical joints	33	2.8293
Valves	5	0.1963×10^{-2}
Filter	2	0.122×10^{-2}
Storage sphere, launch limit switch	1	0.0129×10^{-2}
Total		2.8403

Table 5. Transtage Maximum Propellant Leakage Rate, Main Propulsion System (Based on Specification Rates)

Description	Quantity	Total 30-Day Leakage (lb)	
		Fuel	Oxidizer
Mechanical joints	23	71.14	73.22
Burst disks	4	0.429	0.542
Thrust control valve	2	6.92	8.64
Total		78.49	82.40

3.2.3 Excessive Stage Weight

Certain systems within the Transtage either are not applicable for the Voyager mission because they are peculiar to the Transtage mission or they are excessive in capacity. These items are:

> Propulsion module propellant capability. The present tank capacity is approximately 23,000 pounds of propellant, whereas only approximately 12,000 pounds are required for the Voyager mission, therefore, the tank volume and the pressurizing system volume are oversized for reduced propellant volumes.

- Control module propellant capability. The attitude control system contains approximately 115 pounds of usable propellant which is used for propellant settling and attitude control. Attitude control for the Voyager will be provided by gas jets which are part of the guidance package, but the control module propellant will be necessary for all propulsion system starts, and for providing low shutoff errors for midcourse and orbit trim maneuvers.

- Inapplicable systems. The Transtage vehicle contains all subsystems required for its particular mission in space for periods up to 6-1/2 hours. Those systems which could not be used for Voyager are the separation and destruct system, environmental control system, guidance, power generating, and instrumentation systems.

3.2.4 Diameter Difference Between Transtage and Spacecraft Bus

Since the diameter of the Voyager spacecraft is 240 inches and the diameter of the Transtage is 120 inches, considerable structure will have to be added to accommodate the Transtage vehicle.

3.2.5 Multiple Engines Versus Single Engine

The two thrust chambers which produce a total of 16,000 pounds of thrust are not actually required for the Voyager mission. Elimination of one of the thrust chambers poses problems in the design of a mount structure and a feed system. However, it would tend to give advantages in cost reduction, weight reduction, and increased reliability.

The multiple engine system also poses certain control problems which have to be considered. These are differential starting times and impulses and differential shutdown impulses. The system would, however, have an advantage in that roll moments could be corrected by differential movement of the chambers.

The single engine system would not impose vehicle tumble movements due to differential impulses but it would impart a roll movement to the vehicle which would have to be corrected by an attitude control system. However, changing the Transtage structure to the single engine configuration would require extensive structural redesign and an increase in stage length.

3.3 Recommended Modifications and Feasibility

The preceding section dealt with the problems of the unmodified Transtage in its applicability to the Voyager mission. The subsequent paragraphs consider the problems associated with adapting a modified form of the Transtage vehicle and recommended solutions.

3.3.1 Long Term Storage

In adapting the Transtage to a 1-year journey in space, certain items in the design would require particular attention since as it has previously been stated Transtage space storability is limited to 6-1/2 hours in earth orbit. Those items which would require particular attention include:

- Mechanical joint leakage (gas and liquid)
- Propellant valve leakage (liquid)
- Meteoroid and thermal protection
- Fuel contamination from overboard dump which bleeds control cavity in fuel-actuated main propellant valve
- Cross contamination of propellant through reverse flow past the pneumatic check valves in the pressurizing line. This condition applies to both the main propulsion module and the control module rocket engine feed systems
- Teflon bladders in control module propellant tanks
- Compatibility of tank material.

It has previously been shown that long-term leakage through mechanical joints would be a problem on Transtage. This problem has been recognized by the Martin Company and a solution proposed wherein selected plumbing joints would be brazed or welded to minimize leakage. In addition, test procedures could be changed such that leakage tests would be conducted at higher pressures maintaining allowable leakage rates constant.

Leakage through the main propellant valves, however, poses a difficult problem. The possible solutions would be either to accept the

16.3 pounds/month possible leakage or replumb the engine using explosive valves in a manner similar to the flow schematic (for a single thrust chamber) in Figure 16.

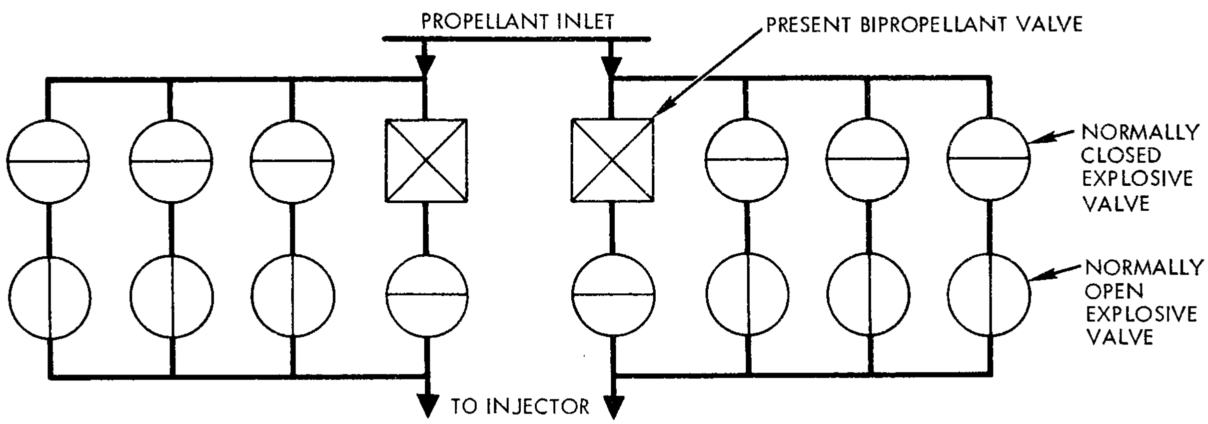


Figure 16. Proposed Revision to Transtage Main Propellant Valves

This plumbing arrangement would seal off the main bipropellant valves with normally closed explosive valves and use the parallel banks of series explosive valves to perform the first three starts required (nominally two midcourse maneuvers and a retrofiring to orbit Mars). Subsequent maneuvers would then be performed by the present valve system whenever the normally closed explosive valves were actuated. The disadvantages to this system are that the present bipropellant valve is mounted atop the injector and a major revision in the mounting arrangement would be required, particularly in view of the fact that the present line sizes on the single engine are 1-3/4 and 1-1/2 inches. Banks of explosive valves this size pose considerable packaging problems. A conventional series design of an integral normally open and normally closed explosive valve would weigh approximately 4 pounds and have a size approximated by a cylinder 8 inches long and 4 inches in diameter. It is conceivable that some reduction in explosive valve size would be possible by allowing some throttling through higher pressure drops associated with smaller valve sizes. However, 1/2-inch squib valves, for example, would cause the engine to be throttled down to 53 psia and operation of the engine at this thrust level would be unacceptable. It would be expected that some tradeoff would be possible such that explosive valve size might be reduced to somewhere around an inch.

It should be mentioned that inclusion of explosive valves in a feed system adds some design problems which are inherently associated with rapid valve actuation. These phenomena are pressure surges at shutdown caused by rapid valve closure and thrust overshoot at start caused by rapid valve opening and the large pressure differences which accelerate flow.

The pressure surges at start and shutdown can be predicted and they generally require increasing wall thickness; the thrust overshoot generally can be overcome by strengthening thrust structure and mounting points. However, the oscillations produced by starting transients sometimes produce an instability within rocket engines which is difficult to predict and correct. The best approach to this problem is to test the engine over a sufficient number of trials to observe stability and then apply conventional techniques in a trial and error fashion.

Meteoroid protection and thermal control devices would be removed from the present systen and a new design proposed, since the present system would be inadequate and the revisions to date consider only 30-day storage. Description of the meteoroid and thermal controls required on the modified system can be found in 5.3 of Section V. This item is not considered to be a problem peculiar to Transtage; however, detailed analysis and verification testing would be required for the final design proposed.

The bipropellant valve exhaust fluid (fuel) is piped overboard through a dump line. This configuration would require the addition of a normally closed squib in order to reduce leakage. Once the system is activated, it is not expected that the fluid ejected would be a problem. Any fluid ejected into space would freeze due to flash evaporation at the pressure of 10^{-13} mm of mercury which exists in interplanetary space, and the addition of a low pressure drop check valve at the end of the line would provide sufficient pressure so that freezing would not occur in the overboard line. An alternate solution to this problem would be to provide an accumulator into which the fluid could be injected.

In each of the rocket engine pressurizing systems, there is a possibility of cross contamination of the propellant tanks because of the common pressurizing system used to pressurize both propellant tanks. In the main propulsion system two series check valves are used in combination with an internal screen and diffuser to prevent reverse flow. It was judged that this system would be sufficient to prevent reverse flow. Quad check valves were not considered because of the low probability of failure of a check valve in the closed position.

The propellant settling rocket engine system however has single check valves in the pneumatic lines to prevent cross-contamination. These would be considered inadequate for the Voyager mission, if the three-ply bladder provided in the fuel tank and the two-ply bladder in the oxidizer tank were retained, because it is a well established fact that diffusion of propellants does occur through Teflon until equilibrium vapor pressure has been established on the other side of the bladder. However, because of this permeability of Teflon and the fact that its ability to maintain structural integrity in the presence of N_2O_4 for the long periods of the Voyager mission has not been adequately demonstrated, conversion of the ullage rocket propellant tanks from bladder expulsion to bellows expulsion is recommended. This results in an improvement in inherent reliability, but it incurs a small weight penalty.

The remaining problem to be considered in long term storage in space is the question raised regarding the possibility of stress corrosion of titanium in nitrogen tetroxide, in particular the alloy 6AL-4V. A review of recent literature has shown that failures have occurred at Bell Aerospace Systems due to stress corrosion. However, tests at the Martin Company at 105,000 psi stress for up to 30 days have not duplicated these results.

The problem is still being evaluated, based on nitrogen oxychloride concentration as the trace ingredient causing the corrosion, with the concentration being a function of the manufacturer. For purposes of this study it was assumed that future results would show that there is no compatibility problem. In the event there is a serious problem a vent system or some means of internal protective coatings would be developed to protect the tanks.

3.3.2 Solar Cell Heating

The present Transtage rocket engine has a radiation skirt that would irradiate the solar cell panels with a maximum heat flux of 220 BTU/min-ft^2. Energy levels of this magnitude incident on the solar cell panel would raise the internal temperature beyond the acceptable operating limits as has been shown in Appendix C, which presents an analysis of the problem. In order to eliminate this incident radiation, the thrust chambers were modified by replacing the radiation skirt with an ablative extension at an estimated weight increase of 118 pounds per thrust chamber. An alternate solution to this problem would be either to store or shield the array. The alternate solutions were rejected because shielding poses problems in raising the radiation skirt wall temperature through reradiation effects and storing the array complicates the maneuvers and adds weight in mechanisms required.

3.3.3 Minimum Impulse Bit

Attaining the minimum ΔV of 1 $\pm$0.1 meters/sec requires use of present 45-pound thruster control system of the Transtage. Use of this system would allow ΔV corrections as small as $1.3 \times 10^{-4} \pm 1.9 \times 10^{-5}$ meters/sec. It would be used for all corrections up to that ΔV resulting from the minimum impulse bit of the main engine, which is approximately 3-5 meters/sec, depending on the gross vehicle weight.

Three midcourse corrections were established as a design criterion and, assuming the main engine system could only produce a minimum ΔV of 5 meters/sec, then maximum single firing duration of the 45-pound modules would be approximately 56 seconds and the total accumulated firing time would be 168 seconds. This duration is well within the 300-second minimum life with sufficient margin remaining to offset the fact that the lifetime is based on a maximum single firing duration of 30 seconds. In adapting attitude control to Voyager vehicle four pitch modules would be used and the yaw-roll module would be eliminated.

3.3.4 Excessive Propellant in Propulsion Module

Since the Transtage propellant tanks are approximately twice the capacity required for the Voyager propellant load, a weight savings was possible by reducing the volume of the propellant tanks and eliminating one of the two 10.25 ft^3 spherical pressurant tanks.

This change can be accomplished conveniently because the propellant tanks are fabricated from separate sections. In order to contain 12,000 pounds of propellants one could shorten the forward oxidizer tank 15.3 inches and the fuel tank 31 inches. This reduction in propellant tank volume not only would reduce tank weight but it would also allow removal of one helium sphere and its attendant hardware. The 18-inch cylindrical spacer between the two 4-inch load rings would be retained for torsional rigidity when adapting the 120-inch-diameter Transtage to the 240-inch spacecraft bus. Prepressurization of the propellant tanks will be required in order to reduce the number of pressurant spheres; however, prepressurization has been a part of the normal operating procedures and it would not represent a change.

3.3.5 Zero-g Starts

In starting the Transtage vehicle in space, where the location of the propellant becomes dependent on geometry and surface tension effects, the yaw and pitch rocket engines are normally fired for 10 seconds producing 1800 lb-sec of impulse, before starting the main engine. Use of this technique ensures that propellant is maintained at the bottom of the tank and it is intended as a secondary precaution even though an elaborate trap and double check valve system is provided at the bottom of the Transtage propellant tanks. In adapting the Transtage system to the Voyager this system would be maintained. The reasons are that the present attitude control system has previously been shown to be required for minimum ΔV magnitude and accuracy requirements and therefore the use of this system requires only additional propellant since the hardware is already available. Alternates to this approach would be either to use the present trap arrangement which would require detailed evaluation, or to provide a start tank system, or to provide capillary devices to orient the propellant. These latter approaches would all be more expensive, more complicated, and would probably result in more weight than the 48 pounds of propellant required for eight zero-g restarts.

Note that the 45-pound thrust chambers are required for starting for all propulsive maneuvers. In addition they are required for the termination of all normal midcourse and orbit trim maneuvers, to keep the shutoff error low enough. However, these engines do not have enough life to produce all the thrust for the midcourse maneuvers of greatest velocity increment. Thus, for at least the first midcourse maneuver, and possibly some orbit trim maneuvers, it would be necessary to operate the main and auxiliary engines simultaneously, with the main engines shut off before the auxiliary engines. Although this complicates the guidance and control functions associated with propulsive maneuvers, it is the only way the Transtage can be used to provide for the Voyager requirements of ΔV magnitude and accuracy for all maneuvers without exceeding engine operating life limits.

3.3.6 Two Engines Versus One Engine

In considering adapting the main propulsion unit to Voyager, both of the 8,000-pound thrust chambers were used along with their mount structure. It would have been desirable, of course, to eliminate one of the chambers in order to reduce the minimum impulse bit and the nonproportional error and also to increase system reliability, since the two-chamber system requires functioning of both thrust chambers. Additional simplification would be achieved in the plumbing and electrical systems and vehicle acceleration, that would be a maximum of approximately 2.5 g's with the 16,000-pound system, would be halved in the 8,000-pound system. This last factor, acceleration, is significant in relation to the inertia loads which would be carried by the truss network supporting various appendages such as antennae and solar cell panels.

A disadvantage of the single engine system concerns the geometry of the present mounting system. In the present system both chambers are mounted interstitially between the two propellant tanks and side loads are transmitted through an interconnected trusswork to the outer shell. Mounting a single engine would require lowering the engine sufficiently so that it could be mounted concentrically to the vehicle centerline. It would then be necessary to design a new thrust mount or adapter system which would be sufficiently stiff to carry the gimballing loads. However, even the single 8,000-pound thrust chamber would have too large a minimum impulse bit to achieve 1 meter/sec minimum ΔV and the maximum acceleration would be 1.2 g's. The advantages of a single engine would include simplicity, cost reduction, and weight reduction, though increased costs associated with a new mount and inlet ducting would be an attenuating factor.

An alternate solution to the problem of large impulse bits could be attempted by lowering the chamber pressure. This would lower the thrust level and in all probability lower the magnitude of the minimum impulse bit. If one assumes a square relationship between minimum impulse bit and thrust level,* then the thrust required would be 1392 pounds, a value which would require a reduction in thrust of 11.5:1 for two chambers and 5.75:1 for a single chamber. Since flow rates for such a system are approximately directly proportional to thrust, system pressure drop would be 0.5 psi for the two-chamber system and 2 psi for the single-chamber system. Obviously at these low pressure drops combustion efficiency would deteriorate and the analogy would break down; however, the point of the argument is the same: that is, pressure drops will vary directly proportional to the square of thrust level; therefore the engine could be sufficiently sensitive or conditionally stable to combustion perturbations such that instability could occur if it were throttled to low thrust levels.

The single engine system was rejected because it was judged that the loss of reliability imposed by abandoning a current development (the

*This can be shown if one assumes constant pulsing specific impulse, constant valve opening time, and that priming of plenum volumes never occurs.

two engine system) and initiating a new development (the one engine system) with the attendant disadvantages of schedule risk and reduced test history outweighed the potential gain in realiability due to increased simplicity.

A blowdown mode of operation was not considered for Transtage even though such a scheme is possible over some finite range of operation. It was assumed a priori that the engine was not throttable without extensive testing and/or modification to the fixed area injector.

3.4 Performance Potential of Transtage

The total usable propellant contained in this modified Transtage propulsion system is 115 pounds in the ullage rocket feed system and 11,761 pounds in the main propulsion.

The performance potential for this system using JPL allocated weights is shown in Figure 17 in which the ideal velocity increment (ΔV) is plotted as a function of propellant consumed. The maximum obtainable velocity increment is 2581 meters/sec for the 1971 Voyager vehicle and 1604 meters/sec for the 1975 Voyager, assuming all the 11,846 pounds of propellant consumed, no capsule separation, and neglecting the loss of 30 pounds of gaseous material which is the product of decomposition of the resin in the thrust chamber ablative material. These performance figures are also computed, predicated on the assumption that propellant used by the ullage rocket system changes the mass ratio insignificantly and that it adds a negligible amount to the velocity increment of the vehicle. The propellant weights required for the mid-course corrections and orbit trim maneuver are 1356 and 300.5 pounds for the 1971 mission and 1885 and 579 pounds for the 1975 mission. Subtracting this propellant from that available and computing the resultant performance available for orbit insertion produces ΔV of approximately 2281 and 1304 meters/sec for the 1971 and 1975 missions, respectively.

If no orbit trim maneuver were conducted before capsule separation, then the propellant required for the 100 meters/sec velocity trim would actually produce 146 and 236 meters/sec for the 1971 and 1975 missions.

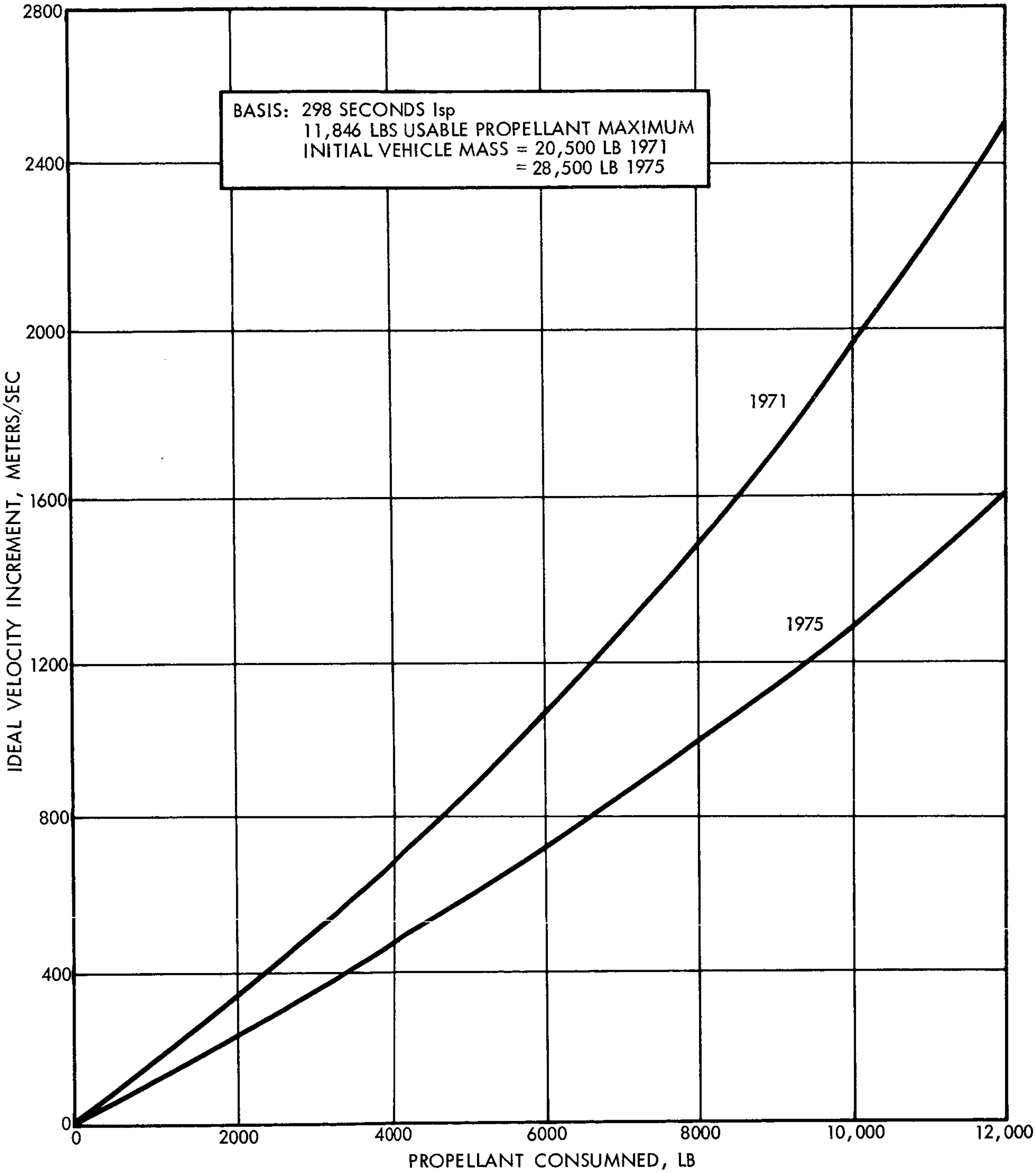


Figure 17. Voyager Transtage Configuration Velocity Increment Versus Consumed Propellant

The minimum ΔV obtainable from the ullage rocket propellant is shown in Figure 18 as a function of propellant consumed, assuming constant vehicle masses of 20,500 and 28,500 pounds. Since a maximum of 42 of the 115 pounds of propellant is set aside to settle the propellant in the main tanks, the difference can be used to provide ΔV either in transit to Mars or in orbit. If it were all used in orbit after capsule separation and without simultaneous main engine operation, then the available ΔV would be 36 and 31 meters/sec corresponding to the 1971 and 1975 missions.

For maneuvers in which the main engines are fired, the minimum magnitude of the velocity increment ranges from 4.8 to 17.8 meters/sec in 1971-73, depending on vehicle weight at the time of the maneuver. (The corresponding figures for 1975-77 weight allocations are 3.4 and 10.3.) Also, main engine firing introduces a nonproportional error due to engine shutoff of ± 1.0 to ± 3.7 meters/sec (3σ), unless the termination of the ullage rockets is after main engine shutoff. (The figures for 1975-77 are ± 0.7 and ± 3.2.)

3.5 Summary

After reviewing the potential problem areas and considering the advantages and disadvantages of alternates and modifications, the system shown in Figure 25 was configured. This configuration consists basically of the main propulsion module of the Transtage with the propellant tanks reduced in volume and off loaded, and with a different environmental control system. The thrust chamber would require modification consisting of removal of the radiation skirt and substitution of an ablative skirt. All joints would be brazed or welded to reduce leakage. The Transtage control module would not be used but the attitude control system would be saved and remounted on the spacecraft structure. Modification to the attitude control system would be made consisting of inclusion of another series check valve in each propellant pressurizing line and substitution of a pitch module for the yaw-roll module. The resulting system would have high performance, reasonable reliability, and moderate cost. Redundancy has been provided in the main pressurizing system through double-series check valves, quad regulator solenoid valves in the main propulsion system, and in the ullage system through parallel regulators,

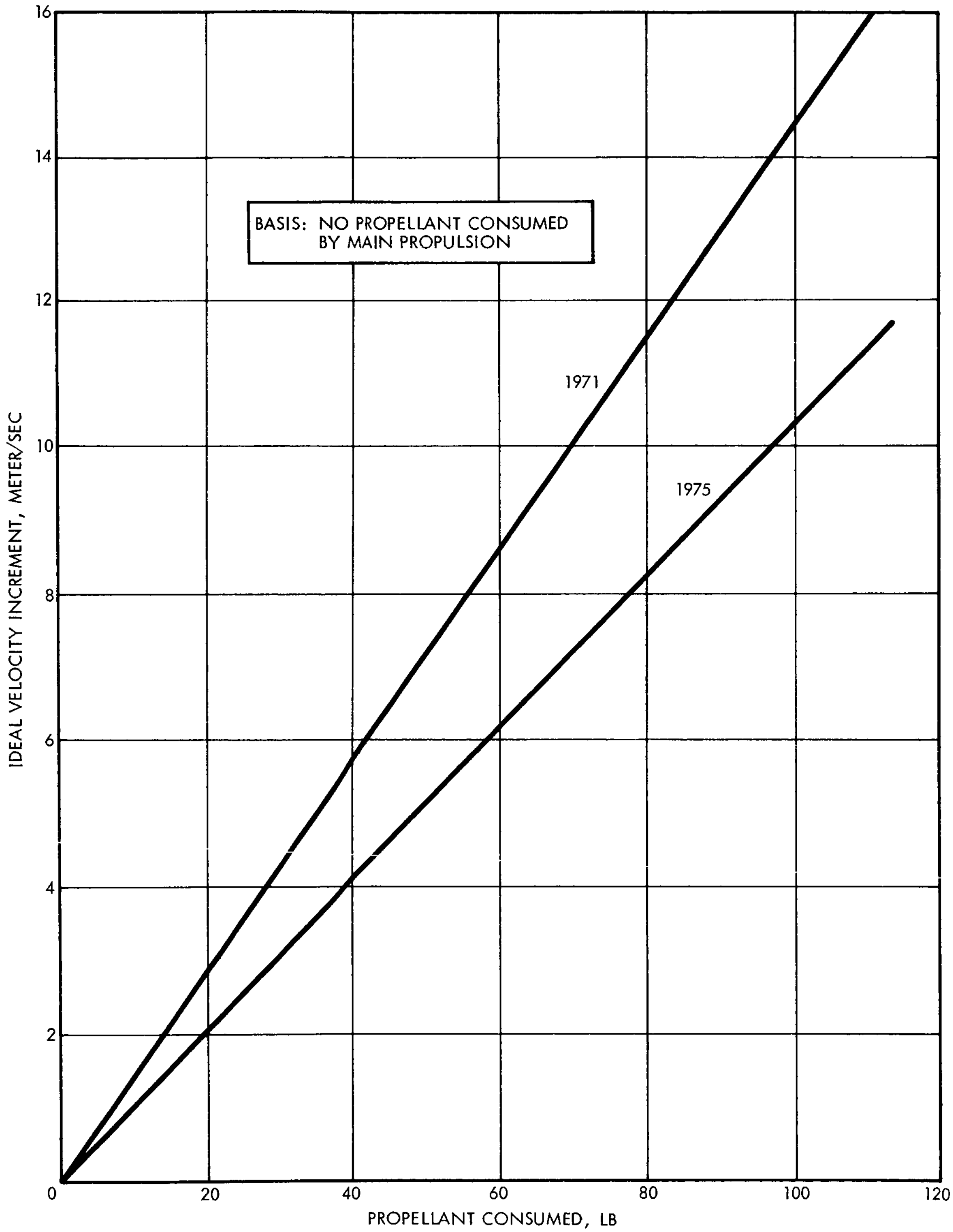


Figure 18. Transtage Configuration Velocity Increment Versus Propellant Consumed by Ullage Rocket System

one of which is activated on failure of the other through means of a pressure switch and three-way squib valve. It is deficient in redundancy and reliability insofar as the main valving on the engines cannot be isolated or provide alternate flow paths and modification to accomplish this would be a significant problem. In addition, both chambers are required for the success of the mission, reducing system reliability from that for a single chamber system. However, the two-engine system will eventually be space-qualified and the single engine would require even more extensive structural development.

The reliability assessment of Appendix A indicates a probability of success of 0.9622 for the main propulsion system and 0.9608 for the auxiliary propulsion system. Factors contributing towards unreliability of the propulsion subsystem are the use of a single main propellant valve in the main propulsion subsystem; single solenoid valves in each propellant line on the ullage rocket engines; and the fact that leakage or failure to open any of the eight single coil solenoid valves in the ullage rocket subsystem would jeopardize the mission through lack of propellant settling and orbit trim capability.

Another factor affecting the reliability is the use of two thrust chambers in the main propulsion system. Failure of either of these rocket engines would cause failure of the mission.

In Appendix B, the cost estimates for the use of the Transtage for Voyager include a propulsion system development cost of $30.2 million, and a propulsion system production cost of $16.2 million for nine units for the 1971 mission. Additional significant costs would be involved over and beyond this if modifications were made in the propulsion system valving, either the main engine or in the ullage rocket system, in order to improve the reliability.

The relatively low cost of the modified Transtage system is due to the use of as much of the existing Transtage hardware as possible. This minimum modification approach reduces the design and development effort normally required in building a propulsion system and a spacecraft.

The development of a modified Transtage will be expedited by the fact that this system has been qualified and is currently going through an

R and D flight test program in which 17 Transtages will be launched into earth orbits. This high level of development is also advantageous because propulsion operational capabilities and characteristics will be well defined.

Flexibility in performing its mission is available in that a variable amount of impulse is available for orbit insertion and small ΔV maneuvers. However, the lack of a common propellant supply limits this flexibility, and indeed requires very careful programming of the propulsive maneuvers to provide starts for all propulsive maneuvers and termination for those propulsive maneuvers with critical shutoff error requirements without exhausting the 115-pound ullage propellant supply or exceeding the 300-second ullage rocket lifetime.

The 23,000-pound propellant capacity of the present Transtage tanks exceed that needed on the present Mars Mission for Voyager. However, it could be an advantage in flexibility in adapting Transtage to Voyager missions to other planets because the original propellant capacity can be restored without exceeding the envelope of 208 inches, by simply using existing Transtage tanks.

4. CUSTOM LIQUID PROPULSION SYSTEM

This section describes a Voyager propulsion system which uses the LEM descent engine, modified as in 2 of this section, and a pressurization and propellant feed system tailored to the requirements of the Voyager mission and spacecraft. The system has two basic advantages over the alternates recommended in the work statement. First, the system will have a more efficient mass fraction than the competing alternates which were designed with propellant and pressurant capacity considerably in excess of Voyager requirements. Second, this system can be efficiently configured into an independent propulsion module. The modularized configuration would permit independent testing of the integrated spacecraft and the propulsion system. This feature has significant implications on the spacecraft development plan. The disadvantages are in the increased development cost and time and the fact that a "performance optimized" liquid system will be less flexible per se than the existing overcapacity competing systems for missions with requirements beyond those of the current Voyager Mission Description.

A drawing of the system and the overall spacecraft configuration is shown in Figure 24.

4.1 Basis for Inclusion in Study

It is recognized that this custom liquid propulsion system is, in a sense, not a contender for selection for the Voyager spacecraft. This is because it is outside the scope of the Task B work statement, but, more importantly, because of the adverse cost and development status, and because of the low priority of any requirement for such improvement in performance capability.

However, it is included in the study for the following reasons:

- It shows what performance is possible within the present state of the art.
- It shows how much benefit (in performance, operating life, shape, modularity, structural weight, etc.) is attained by tailoring a liquid propulsion system to the Voyager requirement, in comparison with the use of presently-developed liquid systems.
- It is useful as an upper limit against which comparisons of other alternates may be made.

4.2 Deficiencies of Existing Systems

In the process of adapting the three specified systems for the spacecraft it was observed that each of the systems had, to varying degrees, certain deficiencies. These deficiencies could be classified into the following categories:

- Modular concepts in the propulsion system were absent such that the spacecraft had to be built around the propulsion system and spacecraft equipment had to be placed according to available areas and volumes.
- The propellant tanks were fixed in shape and mounting arrangement and the pressurizing system and tanks were not optimized for the best weight and volume.
- Thrust level and chamber pressure were fixed and a weight optimization was not used to select the operating points of the engine.

4.3 Description of Custom Liquid Propulsion Subsystems

The custom liquid propulsion system uses the previously described modified LEM descent engine including its integral valving. This engine was selected for this propulsion system because it is clearly in a class by itself for developed engines which are readily adaptable to the Voyager mission, and because development of a new motor would not likely result in any significant performance improvement. The advantages in being able to adjust, or raise, the thrust level means that retromaneuvers can be accomplished with a minimum of gravity losses, and maximum vehicle acceleration can be adjusted according to the limits of the various support structures in the spacecraft. Operation at the lower limit of course negates a requirement for a separate low thrust system for the 1.0 meters/sec minimum ΔV correction during midcourse.

Selection of the LEM engine introduces the disadvantage of fixing the operating points of the propulsion system in terms of propellant combinations, mixture ratio, and chamber pressure. The area ratio of the nozzle was also maintained at its present value of 47.5 even though consideration could be given to trading off increases in nozzle and gimbal system weight against decreases in pressurizing system and propellant tank weight as a function of area ratio. However, previous studies have found that these variables are near their optimum values for stages in the Voyager class and that fairly large variations in these parameters are required before significant performance penalties are incurred.

All other subsystems would also be retained in the custom system; however, repackaging would be accomplished by:

- Redesigning propellant tankage such that a single spherical tank with an internal bulkhead to separate the propellants would contain the entire propellant load. Selection of a spherical shape provides an optimum pressure vessel shape for minimum weight and a good structural shape insofar as elastic instability and load distribution is concerned. Restarts in zero gravity would be provided by a start tank system. Similar to the system discussed for the LEMDS system.

- Redesigning the pressurization system such that all pressurizing system components will be incorporated into modules and storing the helium in two optimum 32-inch spheres. These spheres will be mounted aft of the propellant tankage in special cavities in the honeycomb structure.

- Designing the entire propulsion system as a separate module which would be fitted into the spacecraft as a unit. A honeycomb structure covered with insulation would act as a structure for carrying loads, provide meteoroid protection, and environmental control.

A drawing of the resulting system is presented in Figure 24.

4.4 Performance Potential

The performance potential of the Voyager vehicle using the custom liquid system is shown in Figure 19 in which the ideal velocity increment is shown as a function of propellant consumed, assuming 1415

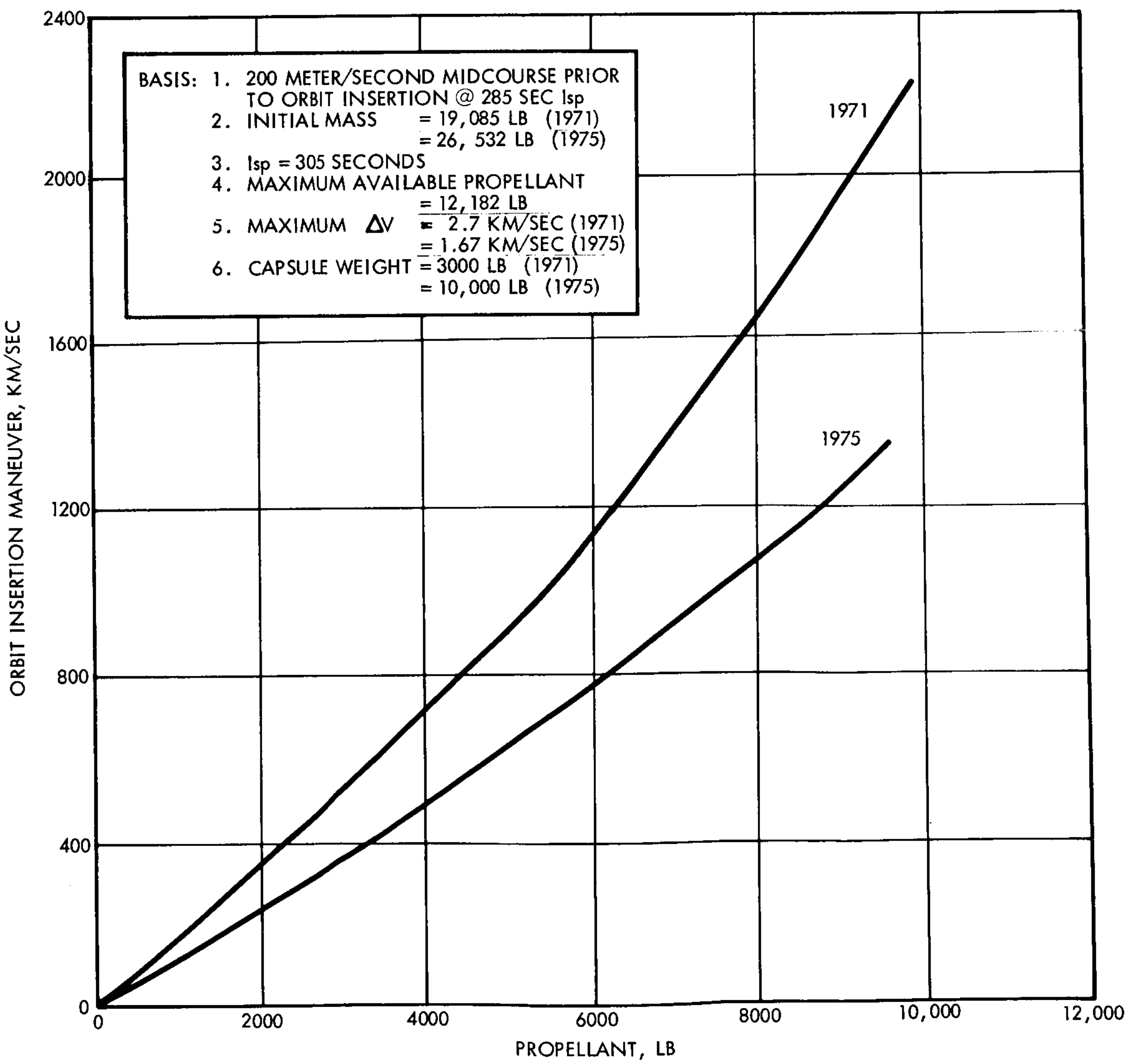


Figure 19. Voyager Custom Configuration Velocity Increment Versus Propellant Consumed

and 1968 pounds of propellant are previously used to provide 200 meters/sec of midcourse correction for the 1971 and 1975 vehicles, respectively. In interpreting the performance of the optional system, it should be pointed out that specific impulse at the high thrust is 305 seconds (orbit insertion) and 285 seconds at the low thrust (midcourse and orbit trim maneuvers).

Total usable propellant for this vehicle is 12,182 pounds, which is nominally distributed as follows:

	Vehicle Configuration	
Function	1971	1975
Midcourse, lb	1,415	1,968
Orbit insertion, lb	9,938	9,620
Orbit trim, lb	829	594
	12,182	12,182

The total ΔV available from this propellant allocation is as follows:

Function	1971	1975
Midcourse (meter/sec)	200	200
Orbit insertion (meter/sec)	2,200	1,347
Orbit trim (meter/sec)	259	100

The velocity increments for orbit trim are calculated, assuming it occurs before capsule separation. (In 1971, all remaining ΔV beyond the desired capability of 2,200 meters/sec for orbit insertion is allocated to orbit trim.) In the event no orbit trim maneuver is conducted before capsule separation, then the propellant so allocated could effect velocity increments of 388 and 242 meters/sec after capsule separation, in 1971 and 1975, respectively. Alternately, if the entire propellant load were to be used for orbit insertion, the corresponding maximum velocity increments would be 2700 and 1671 meters/sec.

5. SUMMARY

The custom propulsion was an idea that grew from the design effort in adapting the various specification propulsion systems to the Voyager vehicle. It was decided to design a system in which these deficiencies were absent in order to provide a broader background against which the three preferred systems could be measured. In doing this, the modified

LEM engine was selected as an optimum prime mover. A modular propulsion system was designed which could be slipped into the spacecraft as a unit. This propulsion system would have a common spherical propellant tank with an internal bulkhead and an optimized stored gas pressurizing system. Modularization of all components would be used in the pressurizing system and the zero-g start tank system.

Because of these modifications, the custom liquid system would have flexible performance in that all propellant is usable in all three phases of this mission. Its main disadvantages would be increased cost in a new propellant tank and helium spheres. Design of new tanks, of course, would add a degree of inflexibility which would limit the applicability of the design to missions requiring fuel capacity beyond the present Voyager requirements, for example, journeys to other planets.

The reliability of this system is basically that of the configuration based on the LEM descent stage. The slightly higher assessment of 0.969 (see Appendix A) results from the reduction in the number of propellant tanks.

The costs associated with the propulsion system of the custom liquid configuration are the highest of all options considered. Appendix B indicates that they include \$44.4 million for development and \$16.0 million for production of nine units for 1971.

V. FLIGHT SPACECRAFT DESIGN CONSIDERATIONS

This section of the report discusses the alternate systems from the point of view of the spacecraft system. It is attempted to preserve the capability of the spacecraft bus identical for all the alternates, in order to achieve a valid comparison. However, the design of the spacecraft bus is necessarily adapted to provide compatibility with the propulsion subsystem of each alternate. For each alternate, the following areas are discussed:

- General features of the spacecraft system design
- Particular geometrical, structural, and configurational aspects
- Weight breakdown (leading to the performance calculations of Section IV)
- Assessment of any comparative reliability degradations of the spacecraft system, due to the environment or geometry imposed by the propulsion subsystem
- Assessment of the comparative cost of the spacecraft system (essentially due to the structure and mechanical subsystems).

1. COMBINATION SOLID ORBIT INSERTION AND LIQUID MIDCOURSE AND ORBIT TRIM PROPULSION SYSTEM

Spacecraft integration and design studies were conducted to evaluate the combination system for the Voyager mission. Physical characteristics of the spacecraft are described, and environmental, operational, reliability, and cost considerations related to the design evolved are discussed in the following paragraphs.

1.1 Voyager Spacecraft with Solid/Monopropellant Module

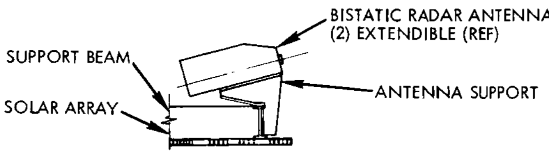
The Voyager spacecraft configuration illustrated in Figure 20 utilizes a modified Minuteman Wing VI Stage II Motor with a monopropellant system for midcourse and orbit trim maneuvers. This system is described in Paragraph 1 of Section IV. The basic frame of the bus structure, of course, is a new development. The solid rocket interfaces with the bus structure at the forward and aft ends of the motor case. A rigid circular frame is secured to the aft end of the motor case and is utilized to complete the frame structure of the truncated equipment compartment when the solid rocket motor is installed. The forward end of the motor case is unrestrained in the thrust direction to allow for case expansion during retro propulsion. Lateral restraint is provided by a ring which is integral with the canted truss frame. Thus, the aft ring attachment serves to transmit the engine thrust and the inertia loads to the space frame of the equipment compartment.

The two monopropellant hydrazine and the two helium pressurization spheres for the midcourse and orbit trim system are supported forward of the solid rocket motor from transverse and intersecting beams. The inertia loads of the monopropellant system are sheared through the webs of the beams and into the aforementioned canted truss-frame structure. The forward and uppermost projections of the space frame are stabilized by a complete ring and the canted truss frame which, in turn, are tied at the emergency separation joint to the aft ring of the aluminum semi-monocoque cylindrical capsule adapter. The four nozzles of the monopropellant system are affixed to the extremities of the system plumbing which is supported by the space frame of the bus structure adjacent to the aft frame of the solid rocket motor case. Lateral restraint is provided by the aft ring of the thrust chamber. Contained within each nozzle are motor driven jet vanes which provide the required thrust vector control.

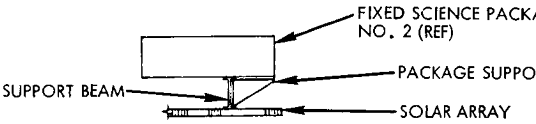
The bus structure is composed basically of a truncated octagonal central equipment bay, integral solar array support frame, and eight truss-type outriggers. This composite serves to react the total planetary vehicle inertia load which is trussed into the vehicle/shroud adapter. The tensile and compressive truss loads are carried into the forward and intermediate frames and are redistributed through the central compartment.

The horizontal members of the outriggers form the plane of the fixed solar array. Auxiliary members complete this frame and provide a rigid platform for the support of the eight identical and fixed solar panels, the PSP (planetary scan platform), the high-medium, and low-gain antennas, the fixed science package, the experiment appendages, the reaction control system, and the capsule/spacecraft antenna.

The geometry of the octagonal equipment compartment was established to satisfy the primary structural requirements and the subsystems

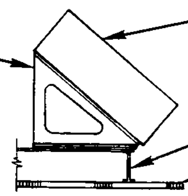


SECTION **N-N**

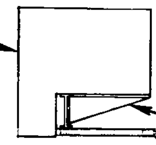


SECTION **M-M**

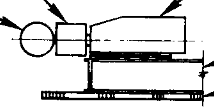
/



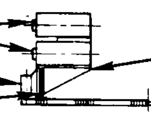
SECTION L-L



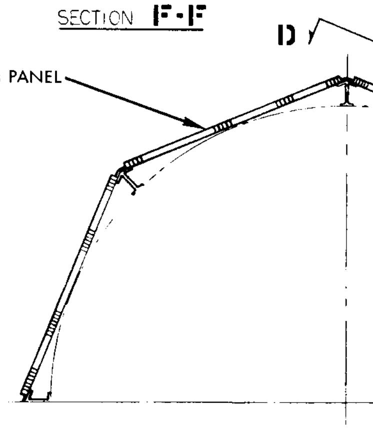
SECTION G-G



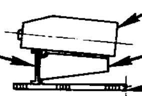
SECTION K-K



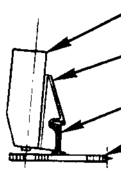
SECTION F-F



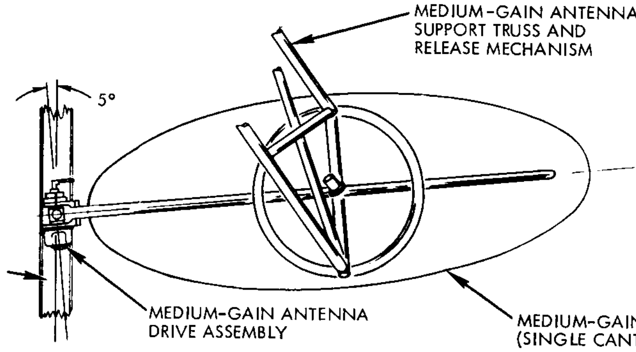
SECTION C



SECTION J-J



SECTION H-H

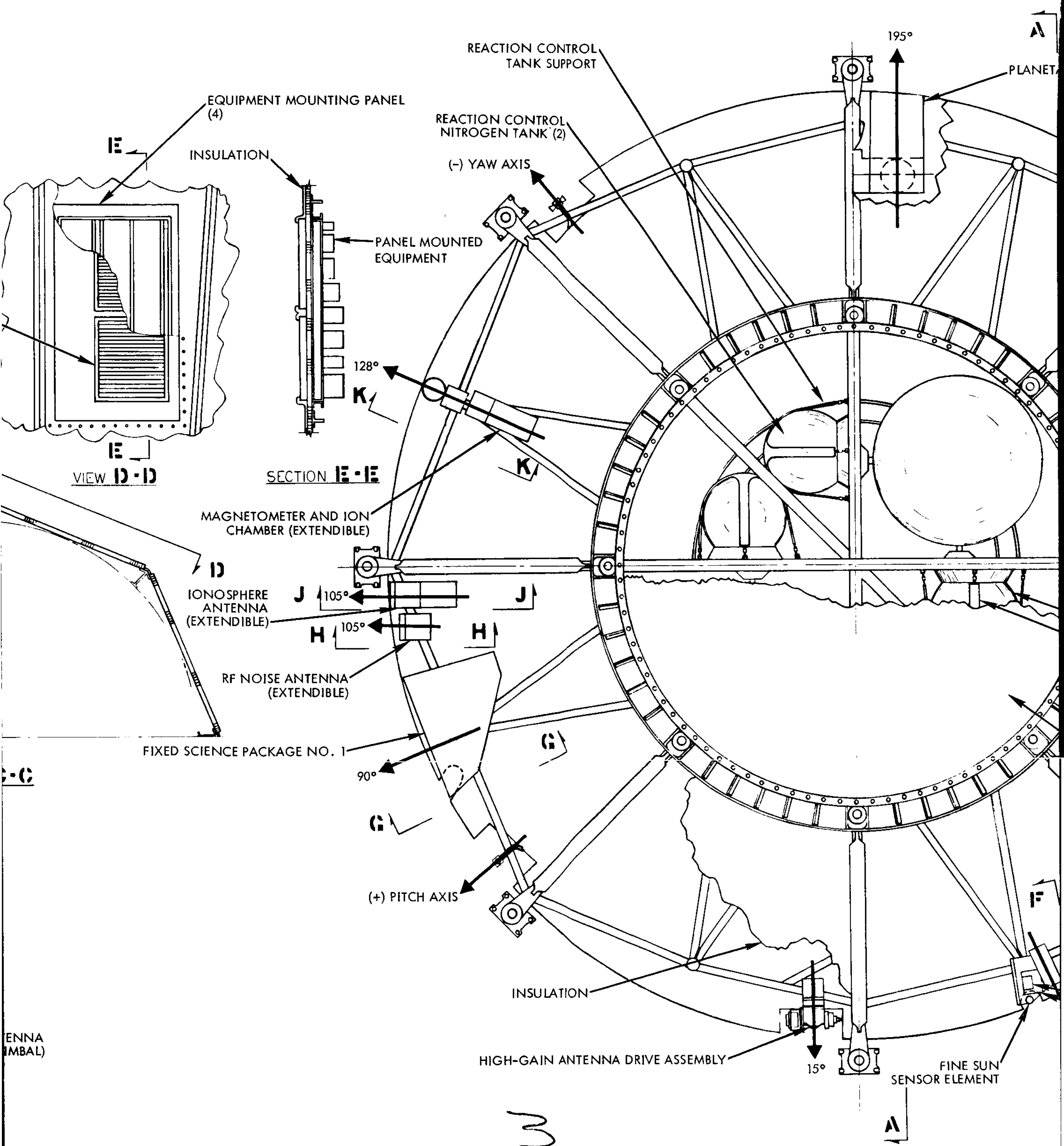


VIEW B-B

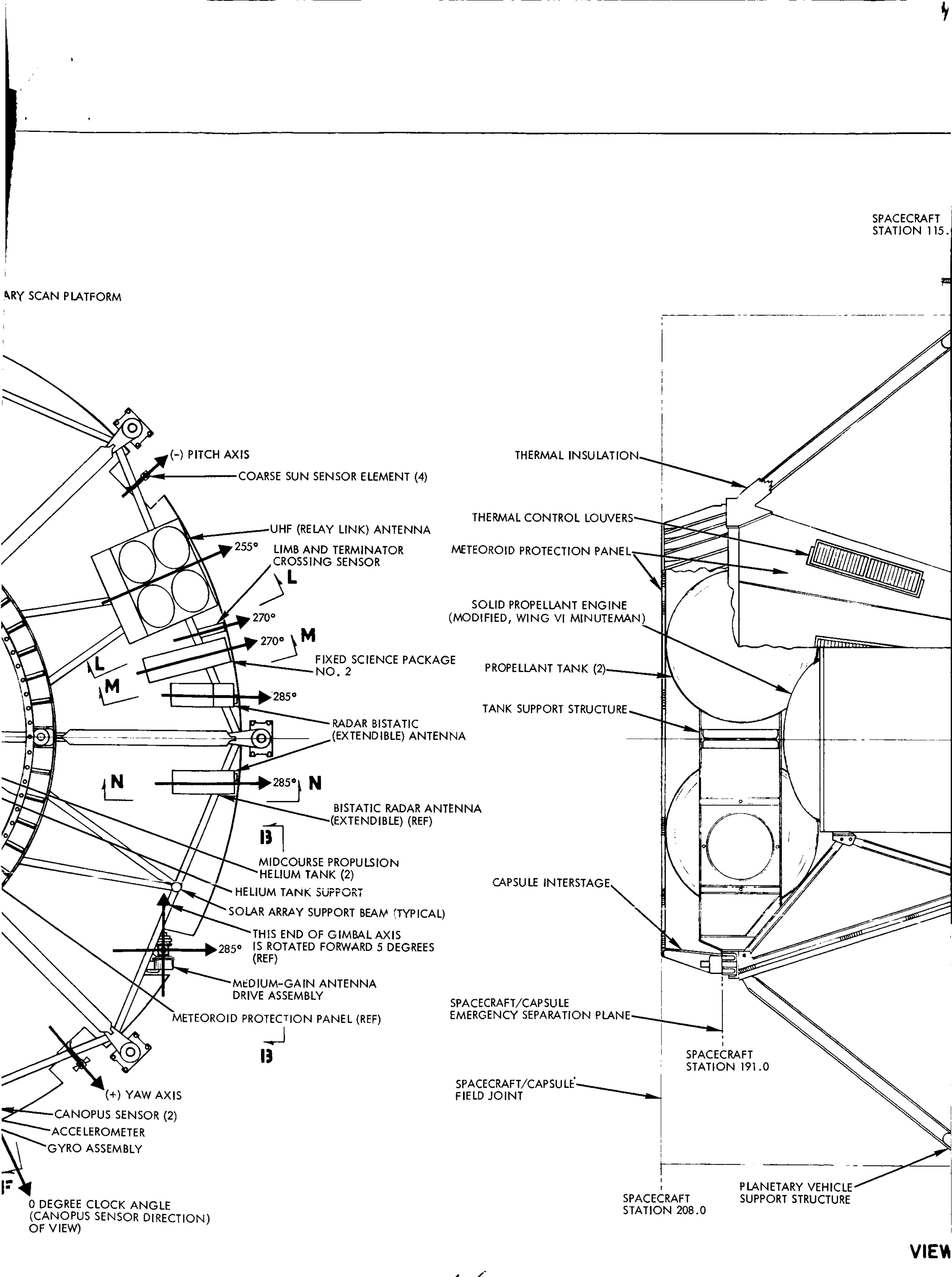
GE

T

2

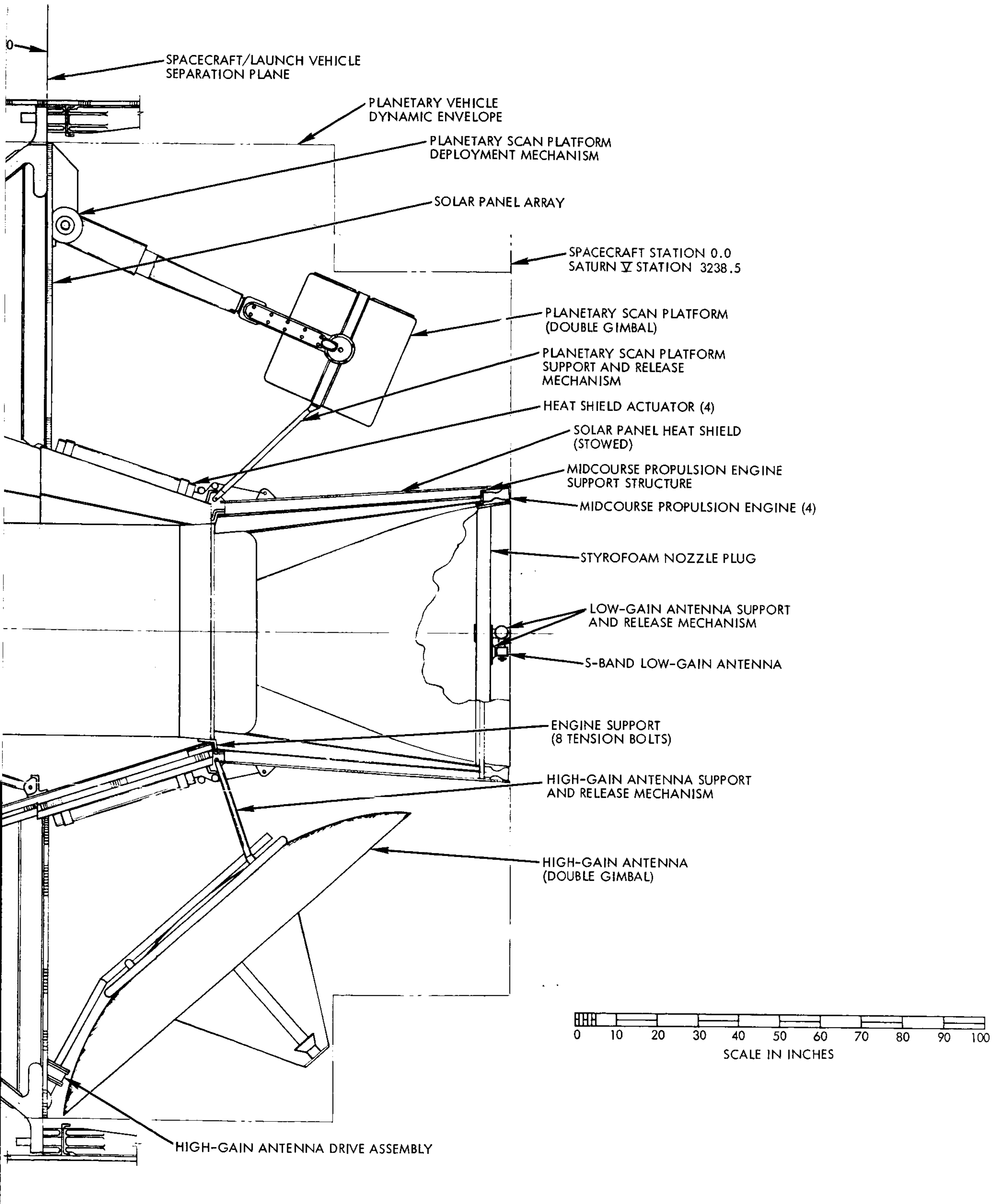

EQUIPMENT MOUNTING PANEL (4)
INSULATION
PANEL MOUNTED EQUIPMENT
VIEW D-D
SECTION E-E
REACTION CONTROL TANK SUPPORT
REACTION CONTROL NITROGEN TANK (2)
(-) YAW AXIS
195°
PLANET
128°
MAGNETOMETER AND ION CHAMBER (EXTENDIBLE)
IONOSPHERE ANTENNA (EXTENDIBLE)
105°
105°
RF NOISE ANTENNA (EXTENDIBLE)
FIXED SCIENCE PACKAGE NO. 1
90°
(+) PITCH AXIS
INSULATION
HIGH-GAIN ANTENNA DRIVE ASSEMBLY
15°
FINE SUN SENSOR ELEMENT
ENNA
MBAL)

3



ARY SCAN PLATFORM
SPACECRAFT STATION 115.
(-) PITCH AXIS
COARSE SUN SENSOR ELEMENT (4)
UHF (RELAY LINK) ANTENNA
255°
LIMB AND TERMINATOR CROSSING SENSOR
L
270°
270°
M
L
M
FIXED SCIENCE PACKAGE NO. 2
285°
RADAR BISTATIC (EXTENDIBLE) ANTENNA
N
285°
N
BISTATIC RADAR ANTENNA (EXTENDIBLE) (REF)
B
MIDCOURSE PROPULSION HELIUM TANK (2)
HELIUM TANK SUPPORT
SOLAR ARRAY SUPPORT BEAM (TYPICAL)
THIS END OF GIMBAL AXIS IS ROTATED FORWARD 5 DEGREES (REF)
285°
MEDIUM-GAIN ANTENNA DRIVE ASSEMBLY
METEOROID PROTECTION PANEL (REF)
B
(+) YAW AXIS
CANOPUS SENSOR (2)
ACCELEROMETER
GYRO ASSEMBLY
F
0 DEGREE CLOCK ANGLE (CANOPUS SENSOR DIRECTION)
OF VIEW)
THERMAL INSULATION
THERMAL CONTROL LOUVERS
METEOROID PROTECTION PANEL
SOLID PROPELLANT ENGINE (MODIFIED, WING VI MINUTEMAN)
PROPELLANT TANK (2)
TANK SUPPORT STRUCTURE
CAPSULE INTERSTAGE
SPACECRAFT/CAPSULE EMERGENCY SEPARATION PLANE
SPACECRAFT STATION 191.0
SPACECRAFT/CAPSULE FIELD JOINT
SPACECRAFT STATION 208.0
PLANETARY VEHICLE SUPPORT STRUCTURE
VIEW

4



5

Figure 20. 1971 Voyager Spacecraft— Solid Configuration

volume requirements superimposed on the large volume occupied by the monopropellant system fuel and pressurization tanks. Subsystem mounting requirements and the mass properties and thermal control constraints dictate the use of four of the eight bays. The face of each bay is split into two panels which are hinged along the outside edges. These equipment or radiation panels support the sensors, batteries, power control unit, tape recorders, science packages, command detector and decoder, and the remaining spacecraft electronic assemblies.

Micrometeoroid protection for the engine systems and sensitive electronics is afforded by a double wall aluminum shield, one face of which serves also as the primary shear panels of the equipment compartment. Immediately aft of the solar array platform, this shielding, in the form of a conical frustum, is utilized to cover the exposed portion of the engine system. The shell is ring-stabilized since it must provide adequate rigidity for the retention and release systems of the high-gain antenna. Of major significance is the incorporation of a 16-petal solar panel protection shield.* This shield is spring loaded in its retracted position adjacent to the nozzles and is supported and hinged from the aft frame of the rocket motor case. Cable cutter initiation, just prior to solid rocket ignition, permits four sets of four petals each to deploy and form a protective cover for the solar array with the ablative surface

*Alternate spacecraft configurations were examined before adopting the one presented, in an effort to eliminate the need for the solar panel shield. These were generally of two classes. The first maintained the same cruise geometry, that is, the solar cells and engine nozzle are both directed toward the sun, parallel to the roll axis. However, the solar cell panels would not be fixed, so that at the time of the orbit insertion maneuver, they could be rotated in place or folded out around the edge of the spacecraft so as to avoid exposure to exhaust plume radiation. In the second class of configuration, a fixed solar panel was oriented in a plane parallel to the thrust/roll axis thereby facing away from the exhaust plume. In this class, the entire planetary vehicle is oriented sideways in cruise, i.e., with the roll axis perpendicular to the sun line. The first class of alternate configuration was rejected because its mechanization was more complex than the chosen solar panel shield. The second class was rejected because the area available for a side-looking array was inadequate for fixed mounting, because the capsule would not be shielded from the sun during cruise, and because not all problems of exposure of the bus to the exhaust plume were needed.

of sandwich structure facing the sun. Thus, a suitable protection against the extreme radiant and convective heating of the exhaust plume is provided. Subsequent to burnout, the initiation of a second set of redundant cable cutters allows the four sets of petals to retract against the engine thrust cone.

To minimize the uncontrolled radiant energy interchange of the main compartment and solar array, an aluminized mylar insulation blanket is provided. This blanket envelopes the exposed truss members, is installed on the back side of the solar array and is tied to the external surface of all micrometeoroid shields. To actively regulate the radiant energy interchange between the main compartment and its environment, a series of bi-metal actuated louvers are attached to each of the aforementioned equipment mounting doors. All other irregular protrusions and seams are suitably insulated to minimize heat leaks.

1.2 Spacecraft Configuration and Geometry Considerations

The various configuration tradeoffs and rationale for selection of the spacecraft design included:

- The configuration design flexibility was limited by the design constraint that the flight capsule must be shielded from the sun during normal flight maneuvers. After due consideration of this constraint and power requirement, it seemed logical to direct all efforts toward establishing an array positioned normal to the planetary vehicle/launch vehicle thrust axis.

- Within the constraints imposed by the 240-inch-diameter shroud envelope and the 90-inch clearance diameter of the Minuteman Wing VI solid propellant rocket installation, it was determined that only 270 ft^2 of fixed solar array could be provided. Although this area does meet the minimum power system requirement of 260 ft^2, it is somewhat less than the design goal of 290 ft^2. In addition, it was necessary to incorporate local cutouts in the array panels to provide clearance for the PSP, high- and medium-gain antennas.

- The 16-petal solar array radiation protection shield provides an acceptable environment for the array; however, the system reliability is adversely affected. Although the proposed system is simple and redundant, failure to open would be catastrophic to the mission, and a random failure when in the open configuration would short the complete solar array. Other concepts considered included articulated solar panels and deployable curtains, but these were rejected due to the mechanical complexity involved and comparatively high development cost.

- The tandem arrangement of the monopropellant tanks and the solid motor was favored over a peripheral arrangement in order to provide an efficient load path between the 10-foot-diameter capsule interstage and the solid motor case and to provide a more accessible installation; however, advantage was taken of the maximum allowable envelope length of 208 inches.

- The limited weight budget for this configuration dictated the utilization of the solar array sandwich panels as primary structure to distribute, in shear, the resultant outrigger lower beam tensile load. The integral structure is dynamically attractive; however, an additional requirement is imposed on the power subsystem. Modularity is also compromised since the bus structure frame is marginally stable without the solar array panels, an approach that necessitates careful and minimum ground handling. In addition, the weight restriction precluded the utilization of redundant aft frames which would be used to thermally isolate the solar array from the bus. The latter approach would reduce induced appendage misalignments and the heater power requirements for temperature control.

- As shown in Figure 20, the monopropellant system tankage protrudes forward into the volume of the capsule adapter which would preclude its utilization for possible ancillary capsule support equipment. In addition, should the available length envelope for the planetary vehicle be reduced, the entire arrangement of subsystems and structure would be grossly affected and would depart from the optimum arrangement considered herein.

- Since the burn-out acceleration of the retropropulsion system approaches 3.0 g's, a programmed appendage articulation would be required to minimize the obvious weight penalty associated with highly loaded cantilevered appendages.

- The modularization of this configuration is somewhat limited in that the complete monopropellant system becomes an integral part of the bus structure; however, all assemblies of this system are readily accessible.

1.3 Weight

A sequential weight summary of the combination solid-liquid configuration for the Voyager mission is presented in Table 6. Also listed in this table are column totals indicating which of the weights are in the spacecraft bus, flight capsule, and propulsion subsystems. These column totals are equal to the weight allocations specified by JPL. The total weights for the spacecraft propulsion and bus are shown as specified

Table 6. Voyager Planetary Vehicle Weight Summary
(Combination Solid-Liquid Configuration)

Item	Capsule weight	Propulsion weight	Bus weight	Total weight
Spacecraft Bus				
Structural and mechanical		1,023	941	1,964
Pyrotechnics			51	51
Temperature control		109	103	212
Radio			126	126
Relay link			25	25
Data storage			72	72
Telemetry			8	8
Command			11	11
Computing and sequencing			36	36
Cabling			229	229
Power			522	522
Guidance and control			232	232
Balance weights			15	15
Contingency			149	149
Spacecraft Propulsion				
Propulsion inert weight		1,264		1,264
Start system inert weight		-		-
Interplanetary trajectory correction inert weight		418		418
Contingency		113		113
Unseparated Capsule Interstage, etc.	250		149	399
Spacecraft Science Payload and Support			400	400
Flight Spacecraft Burnout Weight	250	2,927	3,069	6,246
Flight capsule	2,490			2,490
Jettisoned canister	260			260
Orbit trim propellant (100 meters/sec)		400		400
Planetary Vehicle in Orbit	3,000	3,327	3,069	9,396
Propellant for Mars orbit insertion		9,156		9,156
Inerts expended		267		267
Planetary Vehicle after Interplanetary Trajectory Correction	3,000	12,750	3,069	18,819
Interplanetary trajectory correction propellant (200 meters/sec)		1,681		1,681
Planetary Vehicle Gross	3,000	14,431	3,069	20,500
Planetary Vehicle Adapter				1,500
Total Weight	3,000	17,500		22,000

(17,500 pounds) although the propulsion and the bus weights do not necessarily total to the 15,000 and 2500 pounds independently. This is because of the difficulty in establishing a clearly distinguishable line between the propulsion subsystem and the spacecraft bus.

The combination propulsion system utilizes a modified Minuteman second stage motor with 9156 pounds of solid propellant. Vendor information based on an 11,000-pound system was iterated to obtain the desired propellant loading.

Midcourse correction and orbit trim are provided by a monopropellant system consisting of two 22.8-inch-diameter titanium helium bottles and two 43.6-inch-diameter titanium monopropellant bottles. Propellant settling is maintained by a butyl-rubber positive expulsion system as described in Task A. The weights are based on mass fractions generated during that task.

The spacecraft structural and mechanical subsystems are essentially of the same type construction as in the LEM configuration. However, the combination configuration utilizes the lower member of the outriggers as an integral part of the solar array support structure, and a truss is added to provide further support. Also, the solar array linkage system has been deleted. A blast shield, which protects the solar array from the exhaust plume, consists of a 1-inch-thick core (1.6 lb/ft^3) sandwiched between two 0.01-inch-thick aluminum faces and an operating mechanism.

The only change in the temperature control subsystem is the addition of 0.5-inch-thick refrasil bat on the motor.

The following subsystem weights are assumed to be constant for all configurations and are discussed in Volume 2:

Radio	Command
Relay	Computing and sequencing
Data Storage	Cabling
Telemetry	Power

1.4 Environment Imposed on the Spacecraft and Capsule

The environmental effects of the combination system on the spacecraft and capsule primarily include plume heating of the solar array and acceleration loads imposed by the solid motor during orbit insertion.

1.4.1 Plume Heating

An investigation was conducted to determine the solar array temperature rise and required plume shading area for various distances with several Minuteman motor configurations. (The method used to derive these results is presented in Appendix C.) Table 7 presents the results, which indicate maximum array temperatures will be approximately 1000°F without plume impingement protection. The reason for the high array temperatures, as opposed to temperatures experienced with a liquid propulsion system, is the high temperature radiation from the metal particles at about 3000°F in the solid motor exhaust plume. Figure 21 shows the effect of radial distance as a function of heat transfer rate and shows a comparison of results achieved by several analysis methods at a radial distance of 10 feet.

Table 7. Solid Propellant Motor Plume Heating

Propulsion System	Axial Distance, Ft	Radial Distance, Ft	Incident Heating Rate, Btu/ft^2-hr	Solar Array Temperature at the end of 100-sec firing, °F	Required Plume Shading at q <442, Ft
Modified MM Wing VI	- 6.46 *	10	14600	1150**	38
Modified MM Wing VI	- 9.17	10	9660	900	34
Modified MM Wing VI	- 9.3	10	8300	800	25
Wing VI	-12.7	10	4200	320	25
Wing V (4 Nozzle)	- 6.46	10	9250	850	25

* -6.46 feet axial distance means 6.46 feet above (forward of) the nozzle exit.

** Figures give solar array temperatures at the end of 100-second firing, without the interposition of a protective shield. α/ϵ of the solar array = 1.

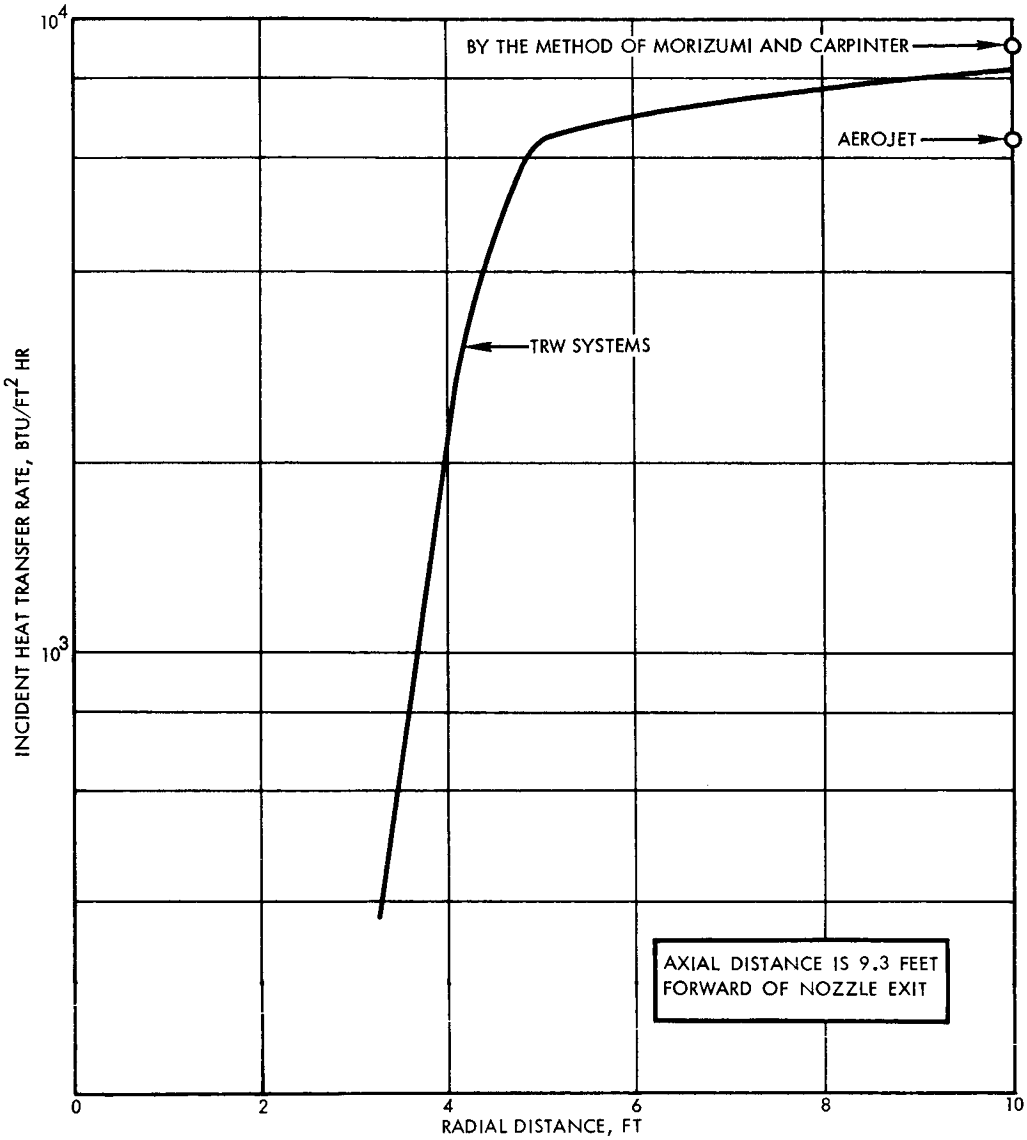


Figure 21. Heat Transfer Rate From Solid-Motor Exhaust Plume Versus Radial Distance

These results emphasized the need for some means of protecting the solar array during motor burn, because of the fact that the solar cell assembly cannot be allowed to exceed 248°F. Since the relative position of the solar array and the solid motor exhaust nozzle was essentially fixed by geometry considerations, it was necessary to provide some mechanically actuated method of protection which, in turn, introduced a degradation in the overall system reliability.

1.4.2 Acceleration Loads

The solid motor imposes an acceleration load of approximately 3.0 g to the spacecraft, due to the inherent burn rate characteristics of the propellant. Although some alleviation of this effect may be achieved through optimal propellant characterization and grain design, the burn action time and average thrust level of the solid motor cannot be appreciably altered. Thus, structural weight and/or system complexity penalties are incurred.

1.5 Other Considerations

Other considerations amenable to tradeoff analyses for the combination system include: operational characteristics, reliability, and cost. Both normal and emergency modes of operation, as well as the combination system-spacecraft and solid motor-monopropellant system interfaces require careful consideration for compliance with Voyager mission requirements.

1.5.1 Operational Characteristics

The monopropellant system may be used with the solid motor to provide additional orbit insertion capability, should the midcourse propellant consumption be less than nominal. Simultaneous firings of both the solid motor and monopropellant system would present thermal control problems, due to the relative proximity of the motor and the propellant supply lines and thrust chamber assemblies of the N_2H_4 system. Another operational feature which was given consideration was thrust termination of the solid motor, should results of mission analysis establish this feature as a mandatory requirement. Available methods for thrust termination, i.e., quenching, venting or nozzle separation techniques, all present additional complexity and tend to induce excessive loads into the spacecraft. Fortunately, it would not appear that thrust termination of the solid is required.

1.5.2 Reliability

A degradation of reliability of the spacecraft functions is caused by exposure to the environment associated with solid-motor firing. Although the formal analyses (based on published failure rate data) generates a factor of 0.9993 for this degradation, attributable to the possibility of heat shield malfunction, a qualitative review, also in Appendix A, results in an estimated value of 0.985. The adjusted probability of success of the combination solid-monopropellant system is 0.949.

1.5.3 Cost

In Appendix B, the following costs are estimated for the spacecraft structure and mechanical subsystems of the combination configuration: development, $11.5 million; and production for the 1971 mission, $12.0 million.

1.6 Summary

The spacecraft design concept for the combination system provides for the utilization of two systems: a monopropellant hydrazine system for midcourse and orbit trim corrections, and a modified Minuteman Wing VI second stage for orbit insertion. Both systems are integrated with the bus structure such that thrust and launch acceleration loads are efficiently distributed for maximum structural efficiency and compatibility with equipment and other spacecraft subsystems installations is provided. The combination system includes features to ensure compliance with operational and functional requirements for electrical power supply and controls, launch operations, thermal control, micrometeoroid protection, and other spacecraft propulsion interface areas. A review of the combination system characteristics with respect to the criteria of 2.5 through 2.8 of Section III would include the following:

- The combination system requires a relatively complex spacecraft installation due to the inherent lack of hardware commonality between solid and liquid propulsion systems. Separate power supply and control functions, thrust vector control, mounting, and instrumentation provisions tend to reduce the modularity characteristics of the system. In addition, exposure to solar array heating and acceleration loads imposed by the solid motor during burn present weight, structural, and reliability penalties to the spacecraft design.
- Reliability and performance characteristics of the flight spacecraft with the combination system should be compatible with Voyager mission planetary quarantine requirements, except for conditions whereby the fixed impulse solid motor provides excessive ΔV for orbit insertion, i.e., should the capsule be jettisoned during interplanetary transfer, the orbit insertion

maneuver must be performed so that excess thrust is consumed. This characteristic of the fixed-impulse solid motor is detrimental from the standpoint of mission flexibility or, if thrust termination is provided, the over-all system reliability is degraded.

- For the combination system, prelaunch handling and operational checkout procedures require that separate storage and shipping units, instrumentation, and logistics functions be provided, thus increasing the overall complexity of prelaunch and launch operations.

Generally, the combination system requires that two separate development programs for the flight hardware and MOSE be conducted and that schedules, specification, and other interface relationships be continuously monitored to assure compatibility.

2. LEM DESCENT PROPULSION STAGE

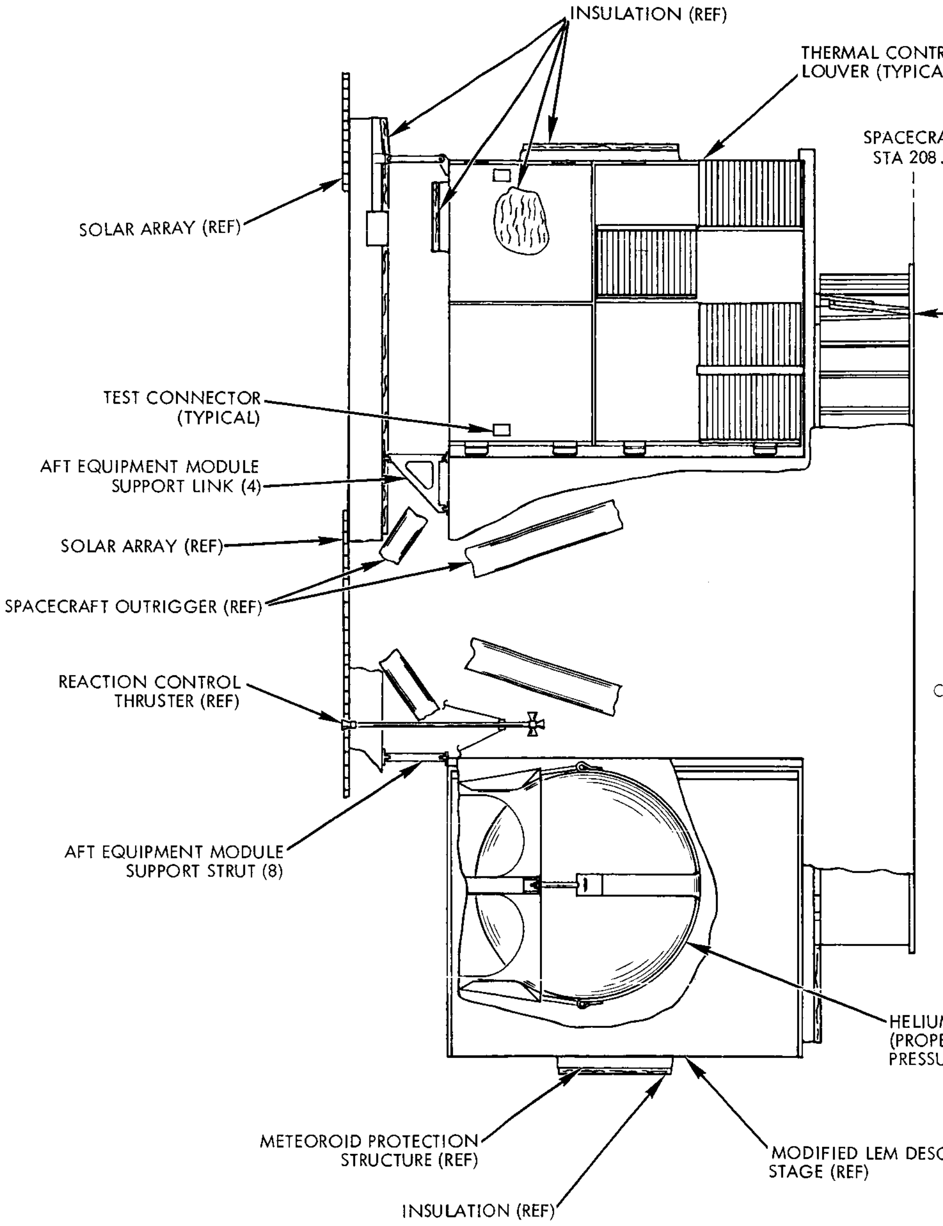
The LEMDS was determined to be readily adaptable to the Voyager spacecraft. Both the physical and functional characteristics of the modified LEMDS are compatible with the specified vehicle equipment and operational requirements. In addition, the LEMDS is amenable to an extended capability spacecraft. Existing stage structure and subsystems are used extensively, and the modifications that are required to accommodate specialized Voyager systems are considered minor in that they do not involve alteration of major components and are implemented using state-of-the-art technology.

The following paragraphs describe and discuss the Voyager spacecraft system based on the LEM descent stage. Geometry and configuration considerations are presented in addition to alternate approaches. Other factors considered in establishing this system, such as weight, imposed environments, reliability, cost, etc., are also discussed.

The discussion here is to the same depth of detail as for the other spacecraft-propulsion alternates. As the Voyager spacecraft based on the LEMD propulsion system has been selected by TRW, a much more detailed description of the spacecraft system and subsystems is the subject of Volumes 1 and 2 of the Task B Study Report.

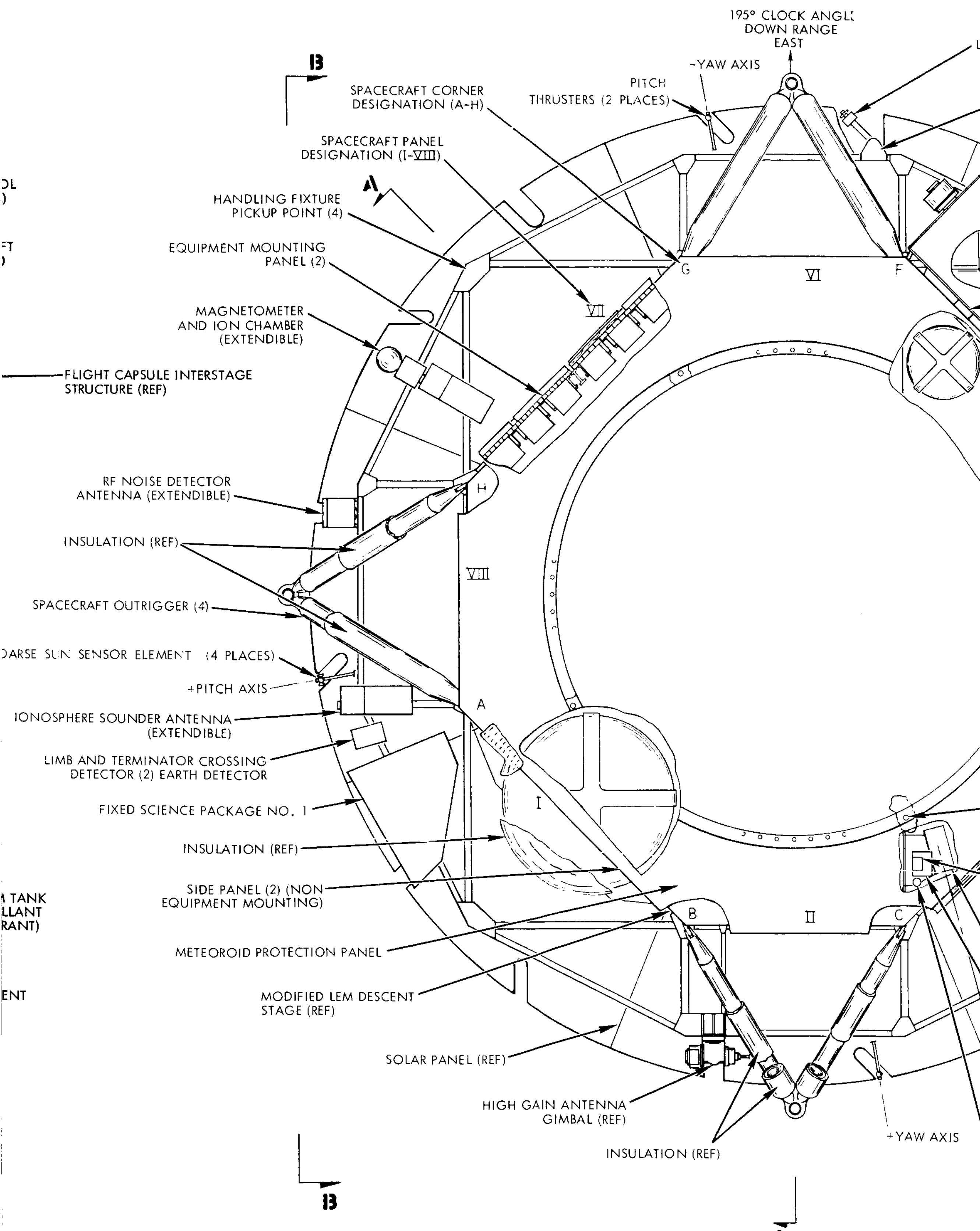
2.1 Voyager Spacecraft with LEM Descent Module

The Voyager spacecraft configuration illustrated in Figure 22 utilizes a modified LEM descent stage to provide the propulsion system

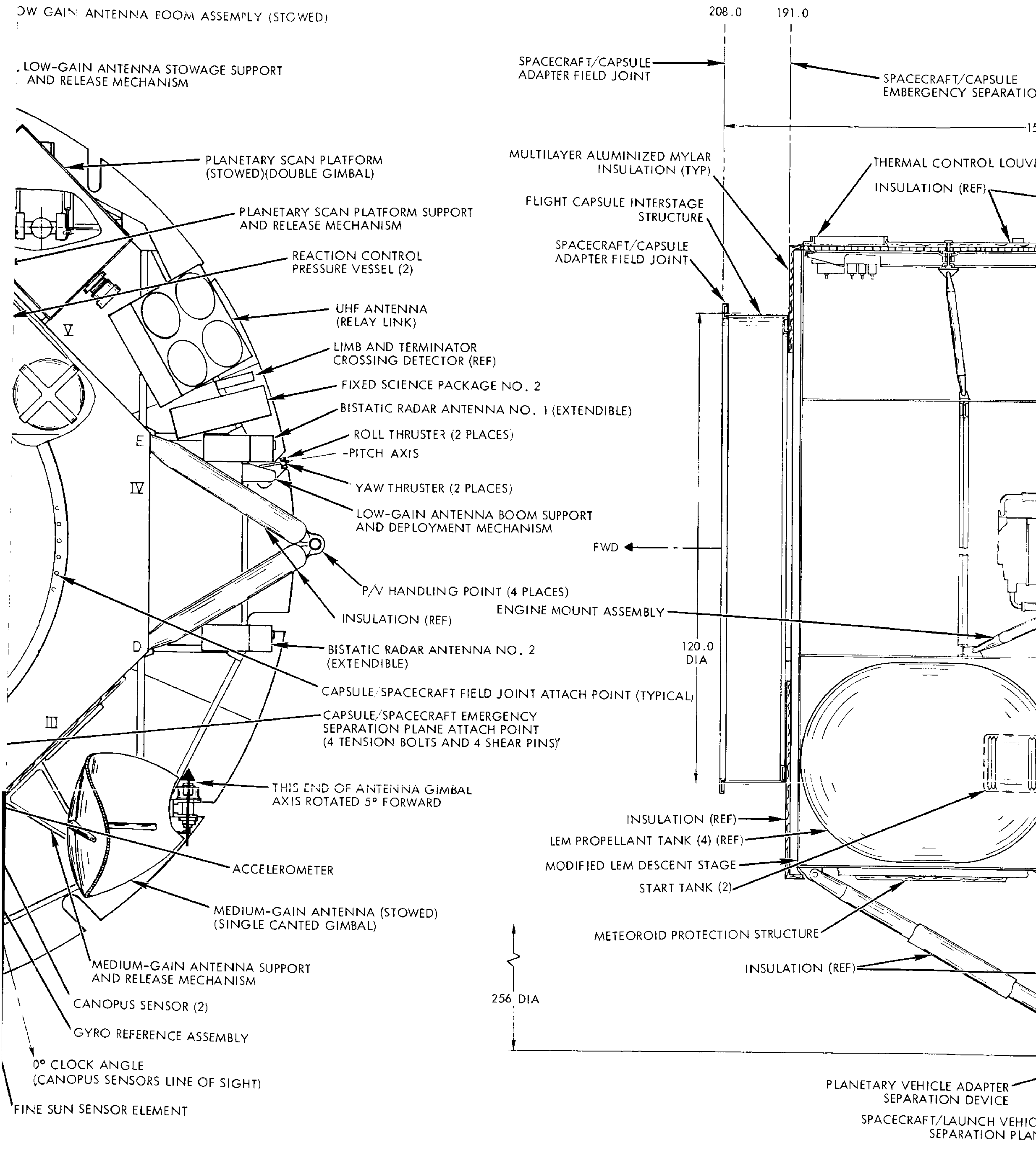

INSULATION (REF)
THERMAL CONTR
LOUVER (TYPICA
SPACECRA
STA 208
SOLAR ARRAY (REF)
TEST CONNECTOR
(TYPICAL)
AFT EQUIPMENT MODULE
SUPPORT LINK (4)
SOLAR ARRAY (REF)
SPACECRAFT OUTRIGGER (REF)
REACTION CONTROL
THRUSTER (REF)
AFT EQUIPMENT MODULE
SUPPORT STRUT (8)
HELIU
(PROPE
PRESSU
METEOROID PROTECTION
STRUCTURE (REF)
MODIFIED LEM DESC
STAGE (REF)
INSULATION (REF)

VIEW B - B

/



195° CLOCK ANGL
DOWN RANGE
EAST
-YAW AXIS
PITCH
THRUSTERS (2 PLACES)
B
SPACECRAFT CORNER
DESIGNATION (A-H)
SPACECRAFT PANEL
DESIGNATION (I-VIII)
A
HANDLING FIXTURE
PICKUP POINT (4)
EQUIPMENT MOUNTING
PANEL (2)
MAGNETOMETER
AND ION CHAMBER
(EXTENDIBLE)
FLIGHT CAPSULE INTERSTAGE
STRUCTURE (REF)
RF NOISE DETECTOR
ANTENNA (EXTENDIBLE)
INSULATION (REF)
SPACECRAFT OUTRIGGER (4)
ARSE SUN SENSOR ELEMENT (4 PLACES)
+PITCH AXIS
IONOSPHERE SOUNDER ANTENNA
(EXTENDIBLE)
LIMB AND TERMINATOR CROSSING
DETECTOR (2) EARTH DETECTOR
FIXED SCIENCE PACKAGE NO. 1
INSULATION (REF)
SIDE PANEL (2) (NON
EQUIPMENT MOUNTING)
METEOROID PROTECTION PANEL
MODIFIED LEM DESCENT
STAGE (REF)
SOLAR PANEL (REF)
HIGH GAIN ANTENNA
GIMBAL (REF)
INSULATION (REF)
+YAW AXIS
G
F
VI
VII
H
VIII
A
I
B
II
C
TANK
LLANT
RANT)
ENT
B
A


OW GAIN ANTENNA BOOM ASSEMPLY (STOWED)
LOW-GAIN ANTENNA STOWAGE SUPPORT AND RELEASE MECHANISM
PLANETARY SCAN PLATFORM (STOWED)(DOUBLE GIMBAL)
PLANETARY SCAN PLATFORM SUPPORT AND RELEASE MECHANISM
REACTION CONTROL PRESSURE VESSEL (2)
UHF ANTENNA (RELAY LINK)
LIMB AND TERMINATOR CROSSING DETECTOR (REF)
FIXED SCIENCE PACKAGE NO. 2
BISTATIC RADAR ANTENNA NO. 1 (EXTENDIBLE)
ROLL THRUSTER (2 PLACES)
-PITCH AXIS
YAW THRUSTER (2 PLACES)
LOW-GAIN ANTENNA BOOM SUPPORT AND DEPLOYMENT MECHANISM
P/V HANDLING POINT (4 PLACES)
INSULATION (REF)
BISTATIC RADAR ANTENNA NO. 2 (EXTENDIBLE)
CAPSULE/SPACECRAFT FIELD JOINT ATTACH POINT (TYPICAL)
CAPSULE/SPACECRAFT EMERGENCY SEPARATION PLANE ATTACH POINT (4 TENSION BOLTS AND 4 SHEAR PINS)
THIS END OF ANTENNA GIMBAL AXIS ROTATED 5° FORWARD
ACCELEROMETER
MEDIUM-GAIN ANTENNA (STOWED) (SINGLE CANTED GIMBAL)
MEDIUM-GAIN ANTENNA SUPPORT AND RELEASE MECHANISM
CANOPUS SENSOR (2)
GYRO REFERENCE ASSEMBLY
0° CLOCK ANGLE (CANOPUS SENSORS LINE OF SIGHT)
FINE SUN SENSOR ELEMENT
V
IV
III
E
D
208.0
191.0
SPACECRAFT/CAPSULE ADAPTER FIELD JOINT
SPACECRAFT/CAPSULE EMERGENCY SEPARATIO
MULTILAYER ALUMINIZED MYLAR INSULATION (TYP)
THERMAL CONTROL LOUVE
INSULATION (REF)
FLIGHT CAPSULE INTERSTAGE STRUCTURE
SPACECRAFT/CAPSULE ADAPTER FIELD JOINT
FWD
ENGINE MOUNT ASSEMBLY
120.0 DIA
INSULATION (REF)
LEM PROPELLANT TANK (4) (REF)
MODIFIED LEM DESCENT STAGE
START TANK (2)
METEOROID PROTECTION STRUCTURE
INSULATION (REF)
256 DIA
PLANETARY VEHICLE ADAPTER SEPARATION DEVICE
SPACECRAFT/LAUNCH VEHIC SEPARATION PLA

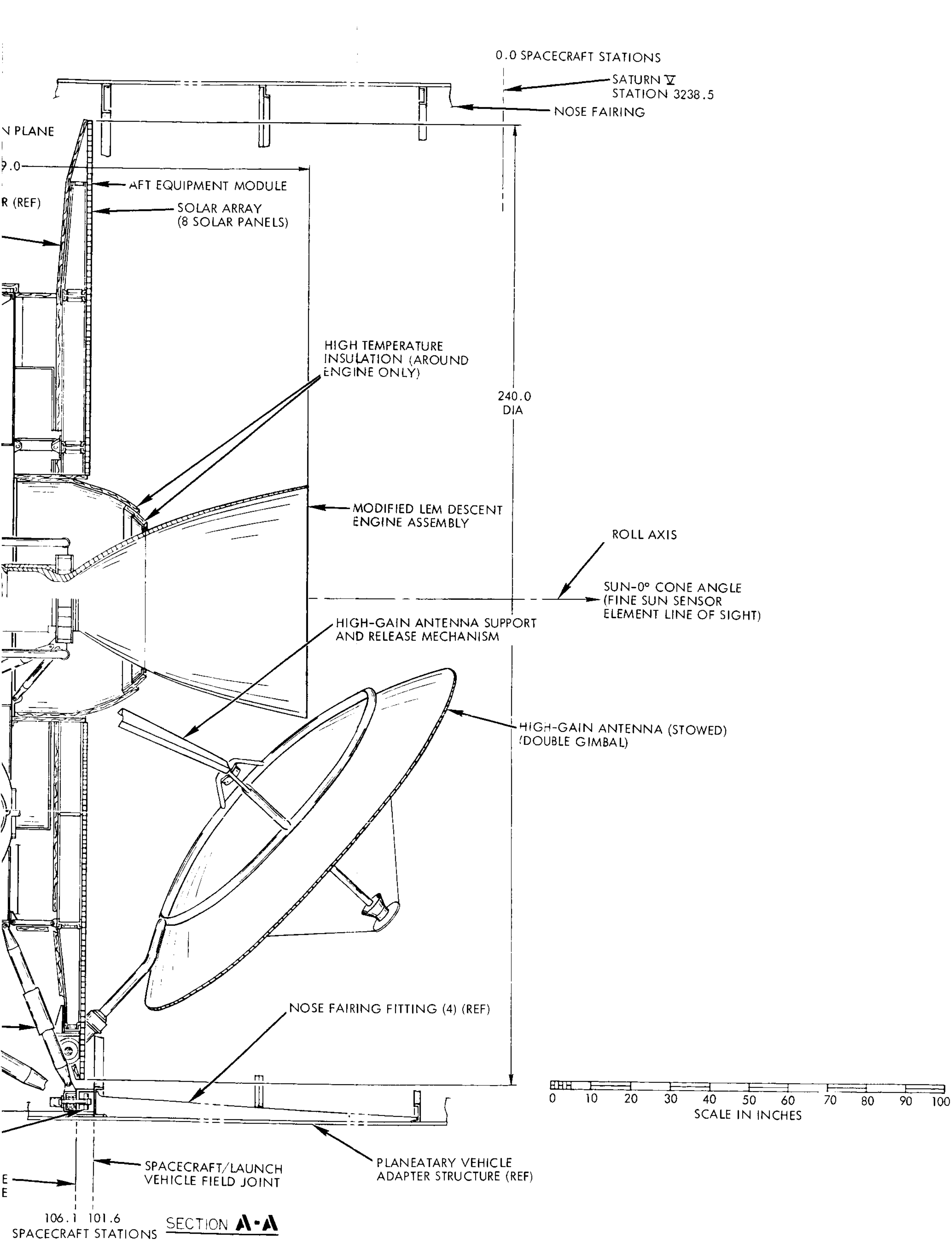


Figure 22. 1971 Voyager Spacecraft–LEM Descent Stage Configuration

and a major portion of the bus structure. The required modifications to and functions of the LEM engine and associated hardware are adequately described in Section IV. The basic frame of the LEM descent stage is used with minor structural modification. This structure consists of two pairs of transverse beams arranged in a cruciform together with upper and lower bulkhead closures. The space between the intersections of the beams forms the 54 by 54 inch center engine compartment. Since the proposed configuration requires that the engine be lowered 36 inches, it is necessary to reinforce the intermediate transverse frame within this compartment to react the radial components of the forward engine mount thrust loads. The aft frame and corner fittings must also be modified to react the radial components of the aft thrust mount loads. The four integral outboard compartments contain the two oxidizer and two fuel tanks.

The external octagonal configuration is completed by the addition of stiffened aluminum skin panels. One of the four corner prismatic compartments contains the single 6 Al-4Va titanium alloy pressure vessel used for helium storage. This 40.9 inch O.D. sphere is supported and preloaded against a scalloped semi-monocoque support structure which is bolted to the aft bulkhead closure. The prismatic compartment diagonally opposite will provide the space for the two 20-inch diameter nitrogen storage vessels for the reaction control system. These vessels, however, will be supported from the aft equipment module rather than from the basic LEM structure. The other two prismatic compartments provide for the support of the Voyager adapted tape recorders, science packages, power equipment, command detectors and decoders, sensors, and the remaining spacecraft electronic assemblies as shown.

The capsule adapter, which is a semi-monocoque titanium cylinder, will extend from the capsule field joint to the emergency separation joint which interfaces with eight machined fittings at the forward bulkhead of the LEM structure. These fittings will be added to the cruciform beam caps in a 10-foot-diameter circle. The capsule inertia loads will be distributed into the transverse beams through existing Z-sections and sheared outboard to the outrigger truss structure. In fact, the inertia loads of all spacecraft equipment are beamed to the outriggers in the same manner.

The existing LEM outrigger structures are replaced with a truss assembly that is tailored to conform with the proposed configuration and to facilitate separation of the planetary vehicle from the launch vehicle adapter. Each of the four outriggers consists of four truss members extending from the planetary vehicle adapter interface inboard to the four outer corners of the cruciform structure. The corner fittings, 16 in total, are redesigned to accommodate the change in magnitude and direction of the tensile and compressive truss loads. The back-up structure for these fittings is capable of sustaining the planetary vehicle loads for the 1971 mission. Minor modification is required for the 1975 mission.

As shown in Figure 22, the basic LEM module is used also to support the aft equipment module which accommodates the PSP (planetary scan platform), the medium-, high-, and low-gain antennas, fixed science package, the reaction control system, the solar array and experiment appendages. Sixteen clevis fittings are added to the underside of the LEM structure at the intersections of the transverse beams and outer bay extremities and interface with the 12-inch links and truss structures which provide the torsional, axial, and lateral support for the aft equipment module. The interface loads are carried into an interlaced arrangement of 6-inch-deep aluminum beams which comprise the frame of the aft equipment module. The beam system geometry readily accommodates the eight identical solar panels, the aforementioned appendages, the reaction control system and science equipment. The solar panels radiate outward from the 62-inch clearance hole, which permits the entire module to be raised or lowered around the engine nozzle extension with or without the solar panels installed.

As mentioned above, two of the four corner prismatic compartments support the major portion of the spacecraft subsystems equipment. The face of each bay is split into four panels which are hinged along the outside vertical edges at the corner longerons of the LEM cruciform structure. These equipment or radiation panels are constructed of a stiffened sandwich consisting of 0.032 aluminum skins bonded to a truss grid core with auxiliary stiffening provided by 3-inch deep hat sections which serve also as the equipment mounting rails. Auxiliary support members are added to support the free edges of these panels. The sandwich panels afford adequate micrometeoroid protection for the internal equipment. However, the basic external webs of the cruciform and prismatic compartments, as well as the forward bulkhead closure, do not afford adequate protection. Therefore, an additional 0.020-inch aluminum skin is required and is separated from the basic existing panel with a 2-inch-thick low density core.

To minimize the uncontrolled radiant energy interchange of the main compartment and solar array, an aluminized mylar insulation blanket is required. This blanket envelopes the exposed truss members, is installed on the back side of the solar array and is tied to the external surface of all micrometeoroid shields. To actively regulate the radiant energy interchange between the main compartment and its environment, two bi-metal actuated louver banks are attached to each of the aforementioned upper equipment mounting doors. The lower equipment mounting panels, all other irregular protrusions and seams are suitably insulated to minimize heat leaks.

Within the constraints imposed by the 240-inch-diameter shroud envelope and the 62-inch clearance diameter for the LEM nozzle extension, it was determined that the design goal of approximately 290 feet^2 of solar cell area could be achieved. However, the optimum cell packaging concept was slightly compromised by the addition of two cutouts in each of the identical solar array panels to provide clearance for the articulation of the high and medium gain antennas. This effect is partially offset by the advantage of cutouts for the mounting and passage of science equipment and sensors.

The geometry of the aft equipment module structure facilitates the installation of the specified appendages and equipment, provides an interface for the ground support equipment, allows for independent module assembly and testing and includes provisions for future growth. These advantages far outweigh the slight weight penalty that may be imposed.

The bus structure is also somewhat less than optimum since there is unused mounting area and volume for equipment; however, the growth potential and the modularity of the LEM frame, the equipment support panels and the outriggers are very attractive and would tend to balance the consideration for optimized structure.

As in all alternate configurations, the 9-1/2 foot high-gain antenna must fold aft in its launch-ready configuration where there is little structure available for support and retention. Support structure is added and extends from the inside edge of the 62-inch-diameter cutout to the tubular frame of the antenna. However, the mast and gimbal assembly of this antenna will easily accommodate the loads imposed during ground handling conditions when unsupported so that the final installation of the antenna can be made after the aft module has been installed.

With the acceleration at the end of the retrofiring limited to 1 g, the majority of the appendages need not be retracted and, therefore, no programmed appendage articulation is necessary with the possible exception of experiment antennas.

When the aft equipment module is raised into position, 12 pins must be installed at final assembly. The four inboard links are readily accessible. The eight outboard links must be installed from an auxiliary platform. This slight disadvantage, however, is far outweighed by the advantages associated with this functional interface. The aft equipment module is able to expand or contract, as the case may be, with no influence on the basic bus structure which, as a result, will minimize, if not eliminate, thermally induced misalignments of the appendages.

2.2 Weight

A sequential weight summary of the LEM configuration for the Voyager mission is presented in Table 8. Also listed in this table are column totals indicating which of the weights are in the spacecraft bus, flight capsule, and propulsion subsystems. These column totals are equal to the weight allocations specified by JPL. The total weights for the spacecraft propulsion and bus are shown as specified (17,500 pounds) although the propulsion and the bus weights do not necessarily total to the 15,000 and 2,500 pounds independently. This is because of the difficulty in establishing a clearly distinguishable line between the propulsion subsystem and the spacecraft bus. The total weight of usable propellant is 11,374 pounds.

The existing LEM descent structure is modified by removing inapplicable items and by providing additional panels and core for

Table 8. Voyager Planetary Vehicle Weight Summary (LEM Descent Stage Configuration)

Item	Capsule Weight	Propulsion Weight	Bus Weight	Total Weight
Spacecraft Bus				
Structural and mechanical		924	628	1,552
Pyrotechnics			51	51
Temperature control		74	111	185
Radio			126	126
Relay link			25	25
Data storage			72	72
Telemetry			8	8
Command			11	11
Computing and sequencing			36	36
Cabling			229	229
Power			522	522
Guidance and control			268	268
Balance weights			15	15
Contingency			135	135
Spacecraft Propulsion				
Propulsion inert weight		2,179		2,179
Start system inert weight		35		35
Interplanetary trajectory correction inert weight*				
Contingency		128		128
Unseparated Capsule Interstage, Etc.	250		149	399
Spacecraft Science Payload and Support			400	400
Flight Spacecraft Burnout Weight	250	3,340	2,786	6,376
Flight capsule	2,490			2,490
Jettisoned canister	260			260
Orbit trim propellant (100 meter/sec)		320		320
Planetary Vehicle in Orbit	3,000	3,660	2,786	9,446
Propellant for Mars orbit insertion		9,654		9,654
Inerts expended				
Planetary Vehicle after Interplanetary Trajectory Correction	3,000	13,314	2,786	19,100
Interplanetary trajectory correction propellant (200 meters/sec)		1,400		1,400
Planetary Vehicle Gross	3,000	14,714	2,786	20,500
Planetary vehicle adapter				1,500
Total Weight	3,000	17,500		22,000

*Propulsion subsystem serves this function also.

meteoroid protection. The additional weight for this protection is 226 pounds. Other weights added to the primary LEM descent structure are miscellaneous supports, latches, hinges, outriggers, aft equipment module, and equipment mounting panels and rails (a total of 628 pounds).

The temperature control subsystem consists of insulation, thermal control louvers, heaters, and thermostats. The existing LEM insulation is removed from all external shell surfaces and replaced by 144 pounds of insulation, 17 pounds of louvers, and 4 pounds of heaters and thermostats.

The LEM engine and valves have been modified for the Voyager mission as follows:

- The nozzle extension and radiation shield were replaced with a radiation skirt which weighs 218 pounds.
- The engine valves have been changed and the electro-mechanical and mixture ratio controls have been removed. Detailed discussion of these items may be found in Volume 2, Section III.

The propellant feed assembly utilizes the existing LEM tankage and plumbing except the flexible propellant lines are shortened. The pressurization system utilizes an ambient helium system with one tank. The main propellant tank supports are the existing LEM supports without modification. Engine and pressurization supports are estimates based on the current location of the LEM engine and one pressurization tank.

The propulsion start system consists of a nonrefillable, 13.6-inch-diameter bellows start tank located one in a main fuel tank and one in a main oxidizer tank. The weight is obtained by ratioing a similar system on the Saturn S-IVB reaction control system, and data generated during Task A.

The interplanetary trajectory correction and Mars orbit trim propulsion utilizes the LEM descent engine, propellant feed system and propellants from the main Mars orbit insertion propulsion subsystem. The propellant weights are based on an $I_{sp} = 285$ sec and $\Delta V = 200$ meters/sec for interplanetary trajectory correction and $\Delta V = 100$ meter/sec for Mars orbit trim with flight capsule attached.

The following subsystem weights are assumed to be constant for all configurations and are discussed in Volume 2:

Radio	Command
Relay	Computing and Sequencing
Data Storage	Cabling
Telemetry	Power

2.3 Environment Imposed on the Spacecraft and Capsule

The modified LEMDS will not present an induced environment problem. The modifications to the stage and engine do not significantly alter the acceptable operation acceleration, vibration, or shock characteristics of the existing LEM. Heating of the spacecraft by the exhaust plume is not considered a problem, (see Appendix C). However, replacing the radiative nozzle by an ablative nozzle extension is necessary to reduce the radiated heat flux from the engine to the spacecraft.

The heat flux, approximately 15,000 Btu/hr-ft^2, from the existing radiation-cooled nozzle extension was found to be potentially detrimental to the solar cell array; therefore, it was necessary to reduce the heat loads. Replacing the radiation nozzle with an all-ablative nozzle lowers the heat flux to an acceptable level of approximately 50 Btu/hr-ft^2.

As the LEM descent engine, as proposed for Voyager, will be limited to a maximum thrust of 7750 lbf, steady-state acceleration will be less than 1g. Thus, for all nondeployable components, spacecraft engine firing imposes a mechanical environment much milder than the launch-induced environment, and therefore is not the designing condition. Even for deployable components, the 1 g limit is less severe than a 1 g ground handling and testing requirement would be. Thus the acceleration due to the spacecraft engine imposes essentially no constraint. The demonstrated stable burning characteristic of the LEMDE also provides an added safety factor in that the vehicle is not subjected to high frequency, destructive vibrations.

2.4 Other Considerations

Other factors such as reliability, cost, schedules, and hardware status were considered in addition to the physical and functional characteristics of the LEMDS. System costs through development, qualification, and production are minimal since a major portion of existing (LEM developed) hardware is retained for the Voyager spacecraft. An exceptionally high reliability potential is also provided by the extensive use of LEMDS man-rated components and the TRW modification ground rule emphasizing minimum risk approaches. Hardware development status, of significant importance in the fixed launch date Voyager program, was also considered prior to selecting the recommended stage modifications, thus insuring development problems would not jeopardize the mission.

The over-all development costs associated with modifying the LEMDS for Voyager application are estimated in Appendix B at $28.1 million, of which $8.1 million is for spacecraft bus structural and mechanical subsystems. This relatively low value results from the use of major components from the LEMDS design which have evolved from the most extensive development efforts; the application of minimum risk approaches in implementing the minor modifications that are required; and the fact that the LEMDS propulsion system imposes minimum environmental requirements on the other spacecraft subsystems. Furthermore, the existing LEMDS geometry is structurally very adaptable to the Voyager space and load path requirements, so that new spacecraft structure development is minimized. The use of the single LEM descent engine (and propellant feed system) for all midcourse, orbit insertion, and orbit trim propulsion functions also tends to reduce hardware costs. Production costs for the 1971 mission include $10.2 million for the spacecraft bus structural and mechanical subsystems and $16.9 million for the propulsion system.

Since the LEMDS and LEMDE are essentially developed systems, their operational characteristics are well defined and only a minimum of preliminary design and development efforts are necessary. A major part of the propulsion funding will be expended in verification and qualification testing and production. The use of the LEMDS is not deemed to result in any reduction of reliability of the spacecraft bus.

Another desirable feature of the LEMDS is its current developmental status and schedule. The initial flight tests and first lunar landing are scheduled for 1969 and 1970, respectively. These dates are compatible with the development requirements imposed by the fixed launch date of the Voyager spacecraft in that the LEMDS propulsion system and structure will be available for early integration and testing with the Voyager subsystems.

The high level of development is also advantageous because propulsion system operational capabilities and characteristics are well defined. This advantage is supplemented by TRW's minimum risk approach toward component selection for the Voyager subsystems and the required LEMDS modifications. These features permit detail design and specification efforts to be initiated early in the program, and provide for firm program planning and scheduling.

2.5 Summary

The LEMDS is readily adaptable to the Voyager mission and a major portion of the LEMDS developed hardware is recommended for use on the Voyager spacecraft. Only minor modifications are required to accommodate the Voyager operational and equipment installation requirements. These modifications employ state-of-the-art technologies, thus providing high reliability at competitive costs and a firm development schedule.

The extensive use of LEMDS hardware also provides a significant degree of mission flexibility. The oversized propellant tanks are filled to only 70 per cent capacity for the basic Voyager mission. Also, the throttling capability of the LEMDE allows a 10:1 range of thrust level to be selected to provide required velocity increments (magnitude and accuracy) while limiting vehicle acceleration loads, without exceeding the engine operational lifetime. In addition, the LEMDS structural configuration is amenable to mounting other scientific instruments and payloads. These features allow a Voyager spacecraft configured from the LEMDS to be used for more ambitious future missions to Mars as well as other planets in the solar system.

The LEMDS is also attractive from a spacecraft design standpoint. Adequate area is available for mounting modularized communications and experiment equipment, solar arrays, and the landing capsule. Mating the modified LEMDS with the Saturn V launch vehicle is relatively simple since the LEM is planned for a Saturn V launch. This feature minimizes the design problems usually associated with spacecraft envelope and launch induced environments.

The modified LEMDS design is also compatible with the planetary quarantine requirement in that the tank pressures are at a reduced level during interplanetary cruise, and the tank pressures and propellants can be vented after the Martian orbit has been established. This minimizes the possibility of a violent spacecraft disintegration from meteoroid impact, tank rupture, etc., causing unsterile fragments to be injected onto a Martian impact trajectory.

Use of the basic LEMDS also allows a major portion of existing LEM support equipment and ground handling procedures to be employed for Voyager spacecraft testing and launching operations. Pressurant and propellant handling equipment used for the Apollo LEM will be satisfactory for the Voyager vehicle configured from the LEMDS and, in addition to equipment compatibility, launch personnel will have acquired valuable hardware experience during the earlier Apollo launches.

3. TRANSTAGE

The Transtage propulsion is readily adaptable to a spacecraft design; overcoming the principal limitations in its application to the Voyager mission is the subject of the modifications proposed in 2 of Section IV. Even so, the probability of success of the propulsion system is inferior to the other alternates, largely because of the extensive use of single solenoid valves and because two thrust chambers constitute the means of prime propulsion. However, it does have high performance, low weight, a high degree of modularity, and it is a developed and qualified propulsion system.

The following paragraphs discuss the effect of this modified Transtage propulsion system on the Voyager spacecraft design and the methodology used in designing a vehicle using this propulsion system.

3.1 Description of Voyager Spacecraft using Transtage Propulsion

The Voyager spacecraft configuration illustrated in Figure 23 utilizes the modified Transtage Propulsion System previously described.

The basic 18-inch long aluminum semi-monocoque cylinder from the Titan Transtage was retained for the integral supporting truss structure for the two engines, the fuel and oxidizer tanks, and the helium pressurization spheres. The propulsion module is attached at its forward extremity to the aft bulkhead frame of the spacecraft bus structure and serves to transmit the engine thrust loads as well as the module inertia loads to the central equipment compartment.

The central equipment bay or compartment which is hexagonal in cross-section, is composed of an integral solar array support platform and six truss type outriggers forming six bays. This composite serves to support the capsule inertia load that is transmitted through the titanium semi-monocoque cylindrical adapter to the forward bulkhead frame of the equipment compartment. The total planetary vehicle inertia load is then trussed into the vehicle/shroud adapter. The tensile and compressive truss loads are carried into the forward and aft bulkheads and are redistributed through the central compartment. The solar array frame also acts as structure that serves a primary function in providing a rigid platform for the support of the six identically fixed solar panels, the four spring-loaded and hinged solar panels, the PSP (planetary scan platform), the high-, medium- and low-gain antennas, the fixed science package, the experiment appendages, the reaction control nozzles, and the capsule/spacecraft antenna. The geometry of the central equipment compartment was dictated not only by the subsystems installation but by the large volume occupied by the propulsion module tanks and pressure vessels and the reaction control system nitrogen storage vessel.

In order to satisfy the center of gravity, environmental control, and subsystem mounting requirements three of these six bays are utilized. The face of each bay is split into two panels which are hinged along the outside vertical edges. These equipment or radiation panels support the sensors, batteries, the PCU, tape recorders, science packages, command detector and decoder, and the remaining spacecraft electronic assemblies.

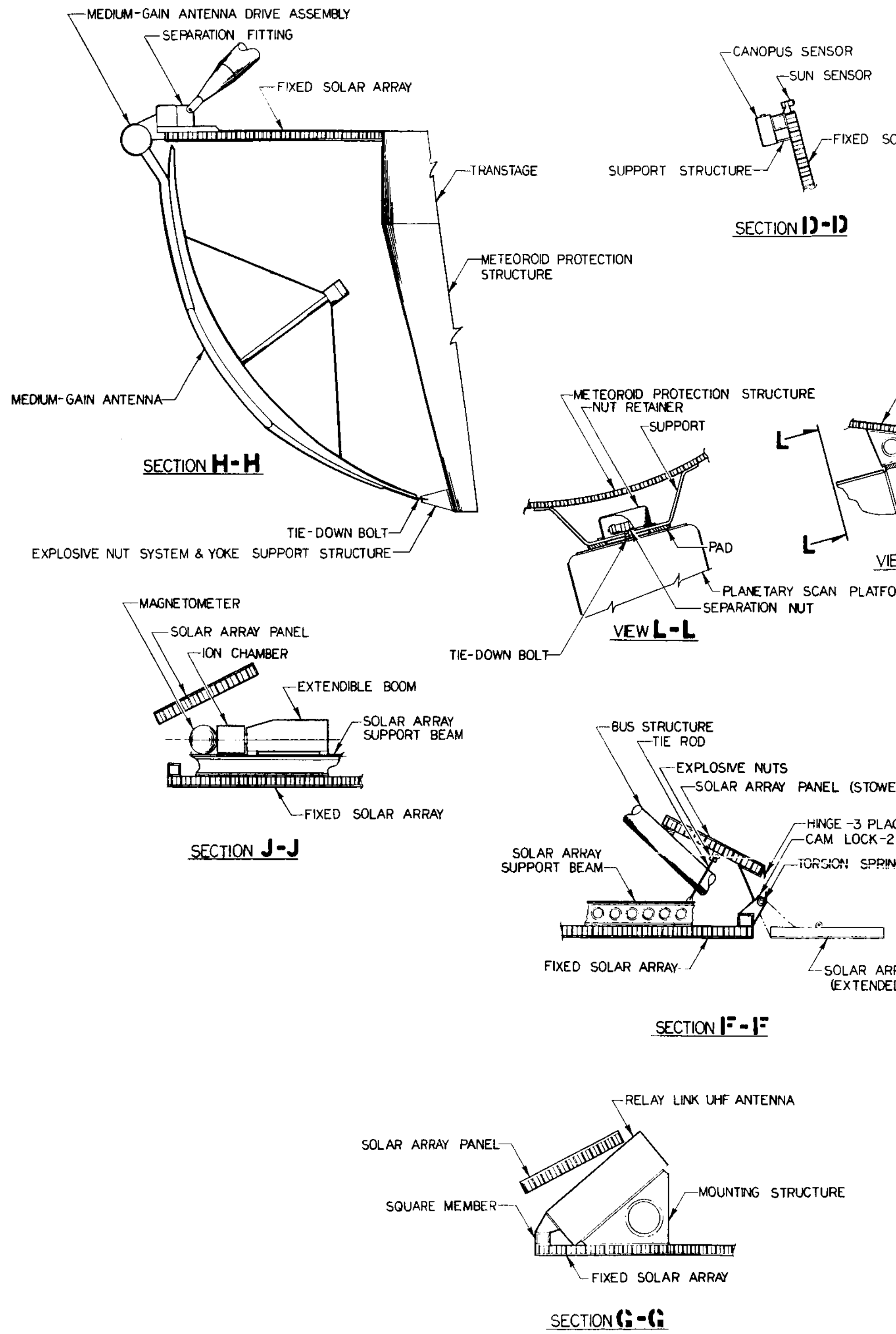
Micrometeoroid protection for the pressure vessels and sensitive electronics is afforded by aluminum shielding which serves also as the primary side shear panels and forward bulkhead stabilizer. An aluminum conical frustum covers the exposed tanks of the propulsion module. This shell is ring stabilized since it must provide adequate rigidity for the retention and release system of the PSP, the high- and medium-gain antennas.

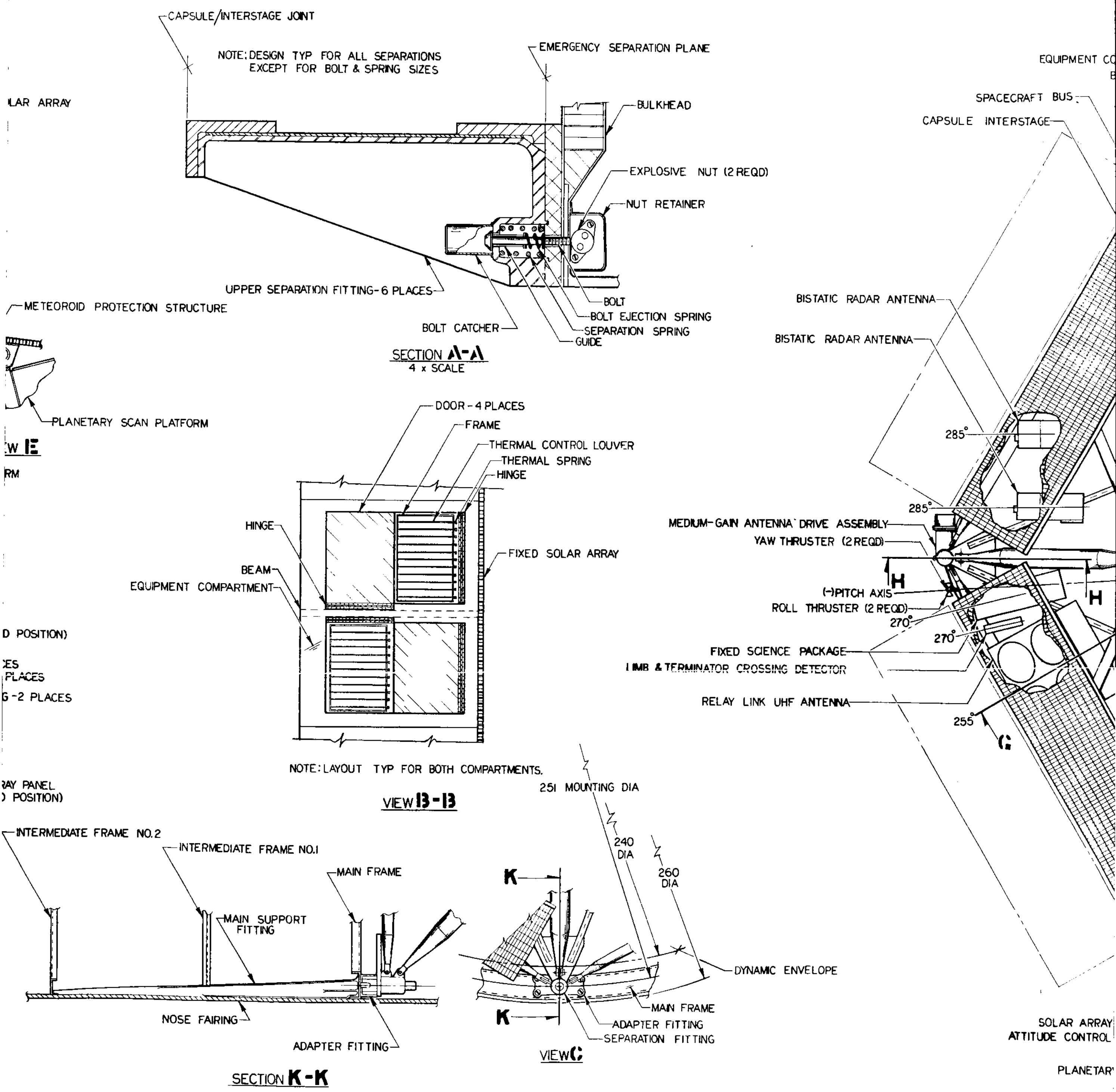
The radiant energy interchange of the main compartment and solar array is minimized by an aluminized mylar insulation blanket. This blanket which envelopes the exposed truss members is installed on the back side of the solar array and it is connected to the external surface of all micrometeoroid shields. Active regulation of the radiant energy interchange between the main compartment and its surrounding environment occurs through a series of bi-metal actuated louvers attached to each of the equipment mounting doors. All other irregular protrusions and seams are suitably insulated to minimize heat leaks.

3.2 Spacecraft Configuration and Geometry Considerations

The configuration design flexibility is limited by the design constraint that the flight capsule must be shielded from the sun during normal flight maneuvers. Consideration of this constraint and the vehicle power requirements, led to the selection of an array positioned normal to the planetary vehicle/launch vehicle thrust axis.

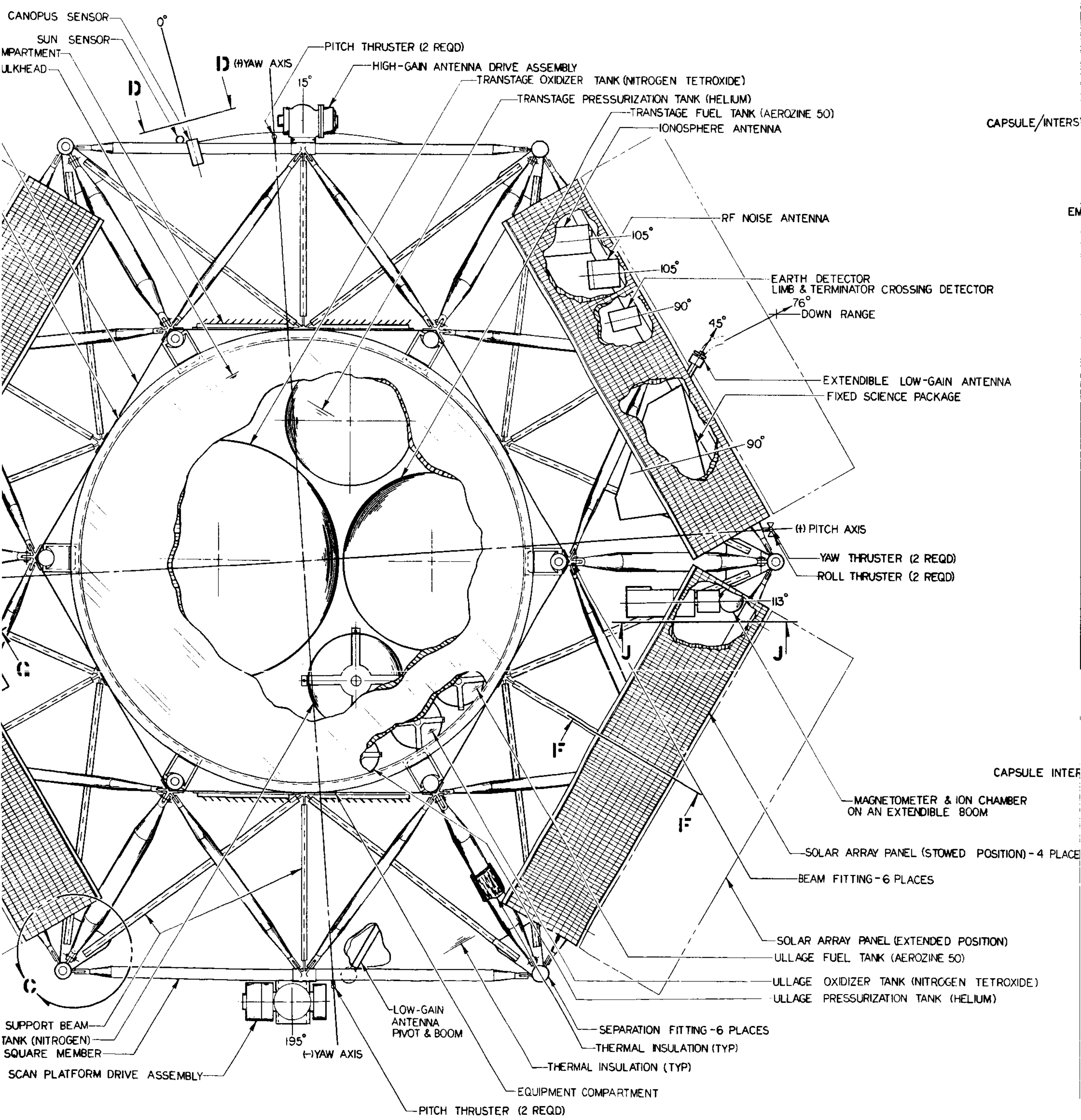
Within the constraints imposed by the 240-inch-diameter shroud envelope and the 120-inch-diameter Transtage propulsion module only 235 ft^2 of fixed solar array area can be provided. In complying with the minimum requirement of 260 square feet, and a design goal of 290 square feet, deployable solar paddles are necessary. These deployable panels led to the selection of the hexagonal platform with hinged panels extending from four of the six sides. This arrangement, however, facilitates the installation of the PSP and spacecraft antennas in that no local cutouts in the solar array are required. It has the disadvantage of reducing the space available for installation and requiring installation of antennas aft


MEDIUM-GAIN ANTENNA DRIVE ASSEMBLY
SEPARATION FITTING
FIXED SOLAR ARRAY
TRANSTAGE
METEOROID PROTECTION STRUCTURE
MEDIUM-GAIN ANTENNA
SECTION H-H
TIE-DOWN BOLT
EXPLOSIVE NUT SYSTEM & YOKE SUPPORT STRUCTURE
CANOPUS SENSOR
SUN SENSOR
SUPPORT STRUCTURE
SECTION D-D
METEOROID PROTECTION STRUCTURE
NUT RETAINER
SUPPORT
PAD
PLANETARY SCAN PLATFO
SEPARATION NUT
TIE-DOWN BOLT
VIEW L-L
L
L
MAGNETOMETER
SOLAR ARRAY PANEL
ION CHAMBER
EXTENDIBLE BOOM
SOLAR ARRAY SUPPORT BEAM
FIXED SOLAR ARRAY
SECTION J-J
BUS STRUCTURE
TIE ROD
EXPLOSIVE NUTS
SOLAR ARRAY PANEL (STOWE
HINGE -3 PLA
CAM LOCK-2
SOLAR ARRAY SUPPORT BEAM
FIXED SOLAR ARRAY
SECTION F-F
RELAY LINK UHF ANTENNA
SOLAR ARRAY PANEL
MOUNTING STRUCTURE
SQUARE MEMBER
FIXED SOLAR ARRAY
SECTION G-G



CAPSULE/INTERSTAGE JOINT
NOTE: DESIGN TYP FOR ALL SEPARATIONS EXCEPT FOR BOLT & SPRING SIZES
EMERGENCY SEPARATION PLANE
BULKHEAD
EXPLOSIVE NUT (2 REQD)
NUT RETAINER
UPPER SEPARATION FITTING-6 PLACES
BOLT CATCHER
BOLT
BOLT EJECTION SPRING
SEPARATION SPRING
GUIDE
SECTION A-A
4 x SCALE
LAR ARRAY
METEOROID PROTECTION STRUCTURE
PLANETARY SCAN PLATFORM
W E
RM
D POSITION)
ES
PLACES
G -2 PLACES
AY PANEL
) POSITION)
DOOR - 4 PLACES
FRAME
THERMAL CONTROL LOUVER
THERMAL SPRING
HINGE
HINGE
BEAM
EQUIPMENT COMPARTMENT
FIXED SOLAR ARRAY
NOTE: LAYOUT TYP FOR BOTH COMPARTMENTS.
VIEW B-B
INTERMEDIATE FRAME NO.2
INTERMEDIATE FRAME NO.1
MAIN FRAME
MAIN SUPPORT FITTING
NOSE FAIRING
ADAPTER FITTING
SECTION K-K
251 MOUNTING DIA
240 DIA
260 DIA
K
K
DYNAMIC ENVELOPE
MAIN FRAME
ADAPTER FITTING
SEPARATION FITTING
VIEW C
EQUIPMENT C
SPACECRAFT BUS
CAPSULE INTERSTAGE
BISTATIC RADAR ANTENNA
BISTATIC RADAR ANTENNA
285°
285°
MEDIUM-GAIN ANTENNA DRIVE ASSEMBLY
YAW THRUSTER (2 REQD)
H
H
(-)PITCH AXIS
ROLL THRUSTER (2 REQD)
270°
270°
FIXED SCIENCE PACKAGE
LIMB & TERMINATOR CROSSING DETECTOR
RELAY LINK UHF ANTENNA
255°
C
SOLAR ARRAY
ATTITUDE CONTROL
PLANETAR

2



CANOPUS SENSOR
SUN SENSOR
MPARTMENT
ULKHEAD
0°
D (+)YAW AXIS
D
PITCH THRUSTER (2 REQD)
HIGH-GAIN ANTENNA DRIVE ASSEMBLY
15°
TRANSTAGE OXIDIZER TANK (NITROGEN TETROXIDE)
TRANSTAGE PRESSURIZATION TANK (HELIUM)
TRANSTAGE FUEL TANK (AEROZINE 50)
IONOSPHERE ANTENNA
CAPSULE/INTERS
EM
RF NOISE ANTENNA
105°
105°
EARTH DETECTOR
LIMB & TERMINATOR CROSSING DETECTOR
90°
76°
DOWN RANGE
45°
EXTENDIBLE LOW-GAIN ANTENNA
FIXED SCIENCE PACKAGE
90°
(+) PITCH AXIS
YAW THRUSTER (2 REQD)
ROLL THRUSTER (2 REQD)
113°
J
J
C
F
F
CAPSULE INTER
MAGNETOMETER & ION CHAMBER
ON AN EXTENDIBLE BOOM
SOLAR ARRAY PANEL (STOWED POSITION) - 4 PLACE
BEAM FITTING - 6 PLACES
SOLAR ARRAY PANEL (EXTENDED POSITION)
ULLAGE FUEL TANK (AEROZINE 50)
ULLAGE OXIDIZER TANK (NITROGEN TETROXIDE)
ULLAGE PRESSURIZATION TANK (HELIUM)
C
SUPPORT BEAM
TANK (NITROGEN)
SQUARE MEMBER
SCAN PLATFORM DRIVE ASSEMBLY
195°
(-)YAW AXIS
LOW-GAIN
ANTENNA
PIVOT & BOOM
SEPARATION FITTING - 6 PLACES
THERMAL INSULATION (TYP)
THERMAL INSULATION (TYP)
EQUIPMENT COMPARTMENT
PITCH THRUSTER (2 REQD)

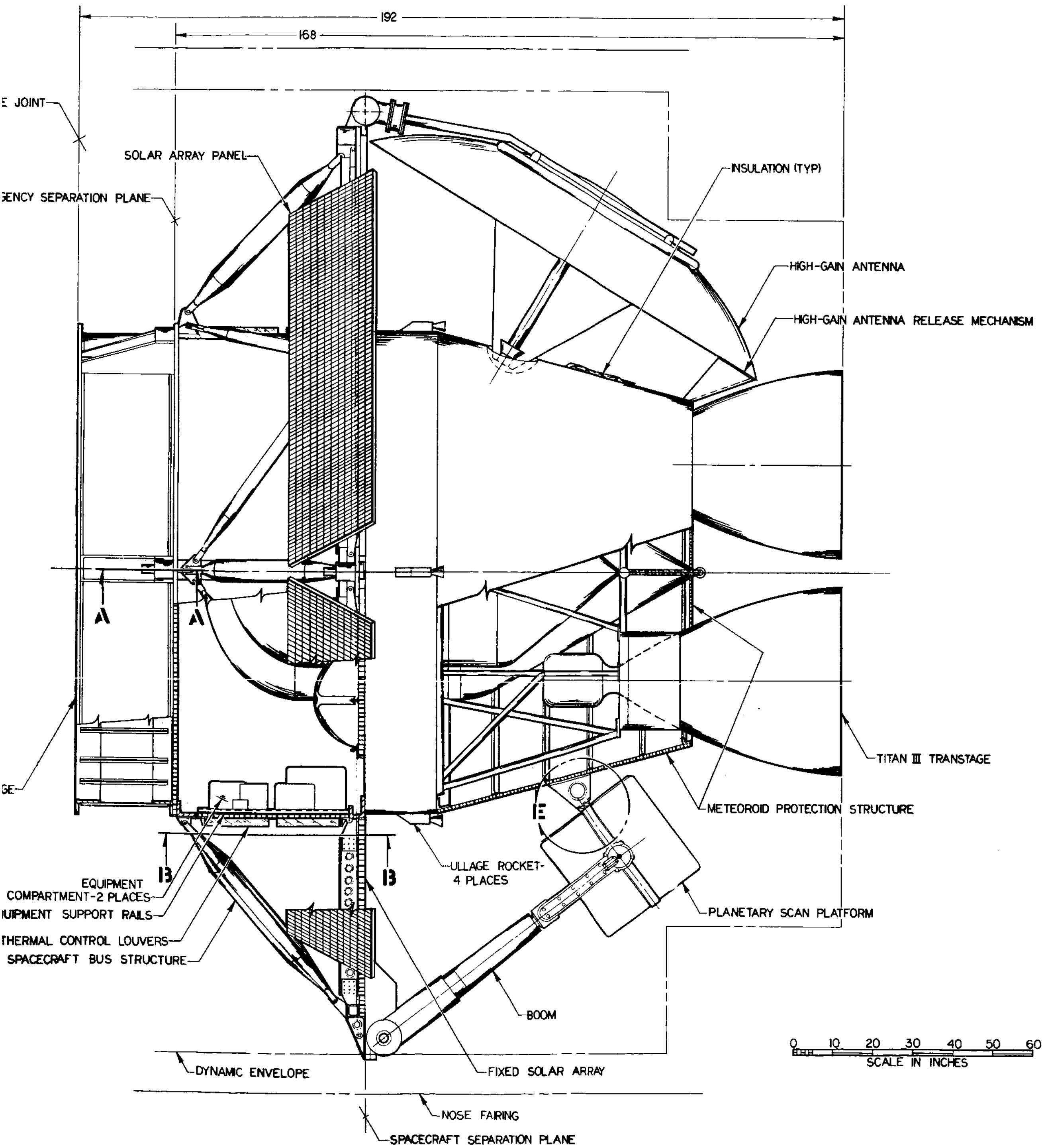


Figure 23. 1971 Voyager Spacecraft— Transtage Configuration

4

of the solar array. In addition, the obstruction of the field of view at this location by the two main rocket engines imposes a constraint on the medium-gain antenna such that the total view angle fails by 30 degrees to encompass the desired 180 degrees.

As shown in Figure 23, the auxiliary fixed solar array platform support members precludes the utilization of the total area of the equipment compartment side panels, and it led to the selection of the double-door equipment mounting concept on three of the six sides of the central bay. Although the compartment accessibility is somewhat limited, the arrangement is acceptable.

Accessibility and capability of facile and expeditious checkout, service and removal of equipment within bus compartment is somewhat restricted by the auxiliary truss members which extend from the forward corners of the bus to the outer frame of the solar array. These truss members could be eliminated if eight rather than six outriggers were utilized; however, a substantial weight penalty would be imposed.

The spacecraft design was expedited by utilizing much of the existing Transtage hardware and fabrication techniques. Thus, the two engines, thrust structure, and the basic 18-inch 10-foot-diameter barrel was retained. To utilize existing tooling, the fuel and oxidizer tanks were merely shortened in length such that existing bulkheads, weldments, and supports could be used.

The tandem arrangement of bus and Transtage structure was favored over a peripheral packaging concept to maximize solar array area and to retain some degree of modularity. The effect on length would be negligible. The required envelope length is 192 inches, which is 16 inches shorter than the maximum allowed.

The adaption of the Transtage propulsion module does provide for a minimum length configuration which would effect a weight savings in shroud structure. The attendant deleterious effects include the following: the higher acceleration associated with two-engine configuration would impose a weight penalty on appendage structure or would require a programmed appendage articulation; the sun incidence angle, if greater than expected, would shadow the fixed solar array which would affect the solar cell power output; the large envelope of the propulsion module constrains the arrangement of spacecraft antennas.

3.3 Weight

A sequential weight summary of the Transtage modified for the Voyager mission is presented in Table 9. Also listed in this table are column totals indicating which of the weights are in the spacecraft bus, flight capsule, and propulsion subsystems. These column totals are equal to the weight allocations specified by JPL. The total weights for the spacecraft propulsion and bus are shown as specified (17,500 pounds) although the propulsion and the bus weights do not necessarily total to the 15,000 and 2,500 pounds independently. This is because of the difficulty in establishing a clearly distinguishable line between the propulsion subsystem and the spacecraft bus.

The Transtage propulsion subsystem inert weights were determined by eliminating inapplicable subsystems from the JPL data ("Design Data for Candidate Voyager Spacecraft Propulsion Systems," dated November 12, 1965). The propellant tanks were shortened by removing 15.3 inches from the oxidizer tank and 67.2 inches from the fuel tank.

The new tank weights were obtained by a direct volume ratio. Propellant plumbing remained constant. The pressurization system was altered by reducing the pressurant, bottles and plumbing by a factor of two. An ablative skirt was added to each engine increasing the weight by 118 pounds for each thrust chamber.

Spacecraft bus structure and mechanical subsystems are essentially the same type construction as the configuration based on the LEM descent stage. The Transtage configuration contains additional latches, hinges, and pyrotechnics required for the deployable solar paddles.

The following subsystem weights are assumed to be constant for all configurations and are discussed in Volume 2:

Radio	Command
Relay Link	Computing and Sequencing
Data Storage	Cabling
Telemetry	Power

Table 9. Voyager Planetary Vehicle Weight Summary (Transtage Configuration)

Item	Capsule Weight	Propulsion Weight	Bus Weight	Total Weight
Spacecraft Bus				
Structural and mechanical		759	587	1,346
Pyrotechnics		-	58	58
Temperature control		58	110	168
Radio			126	126
Relay link			25	25
Data storage			72	72
Telemetry			8	8
Command			11	11
Computing and sequencing			36	36
Cabling			229	229
Power			522	522
Guidance and control			251	251
Balance weights			15	15
Contingency			130	130
Spacecraft Propulsion				
Propulsion inert weight		1,928		1,928
Start system inert weight		38		38
Interplanetary trajectory correction inert weight*				
Contingency		112		112
Unseparated Capsule Interstage, etc.	250		149	399
Spacecraft Science Payload and Support			400	400
Flight Spacecraft Burnout Weight	250	2,895	2,729	5,874
Flight capsule	2,490			2,490
Jettisoned canister	260			260
Orbit trim propellant (100 meters/sec)		295		295
Planetary Vehicle in Orbit	3,000	3,190	2,729	8,919
Propellant for Mars orbit insertion		10,216		10,216
Inerts expended		30		30
Planetary Vehicle After Interplanetary Trajectory Correction	3,000	13,436	2,729	19,165
Interplanetary trajectory correction propellant (200 meters/sec)		1,335		1,335
Planetary Vehicle Gross	3,000	14,771	2,729	20,500
Planetary vehicle adapter				1,500
Total Column Weight	3,000	17,500		22,000

*Propulsion subsystem serves this function also.

3.4 Environment Imposed on the Spacecraft and Capsule

The environments imposed by the propulsion system on the spacecraft will not be detrimental to the spacecraft.

Detailed analysis of the heat transfer effects on the spacecraft can be found in Appendix C. It is shown there that the radiation cooled nozzle extension would have to be replaced by an ablative nozzle in order to reduce incident radiant flux from 220 to 1 Btu/ft^2-min. Plume heat transfer was also shown to be confined to strictly radiation, since jet expansion is not expected to cause impingement on the solar cell or other appendages.

Acceleration levels will be a maximum of 2.7 g's (1971 mission) and 2.3 g's (1975 mission) after capsule separation. These high accelerations will require either additional strengthening of deployed and articulated components to support the inertia loads, or a programmed articulation of such components to an insensitive orientation while the main engines are fired.

Data was not available on actual vibratory and shock loads imposed by the engine; however, Transtage components' vibration and shock specification values are reported in the JPL specification. Loads of the magnitude reported there are not expected to be a structural problem on this spacecraft.

3.5 Other Considerations

The reliability of the Transtage propulsion system as it would be adapted to the Voyager mission is 0.924. The details of the analysis from which this reliability was estimated are presented in Appendix A. The reliability is predicted for a total mission time of approximately 6 months with 469 seconds of main engine firing and approximately 200 seconds of ullage rocket engine system engine firing. Individual reliabilities for these propulsion systems were 0.9622 and 0.9608, respectively. The only reliability effect felt to be imposed on the spacecraft is that associated with the release and deployment of solar cell panels, assessed at 0.9990. (See Appendix A.)

The estimated comparative costs (Appendix B) of the spacecraft bus of the Transtage configuration are modest: bus structural and mechanical subsystem development is $10.1 million and production for the 1971 mission is also $10.1 million.

3.6 Summary

In adapting the Transtage propulsion system to a Voyager spacecraft, the result vehicle would have a weight of 17,500 pounds consisting of a 2727-pound bus and a 14,771-pound propulsion system of which 11,846 pounds would be usable propellant.

The length of the spacecraft would be 192 inches, or 16 inches shorter than the maximum permitted value, 208 inches. In the event additional propellant were desired, the propulsion tanks could be expanded back to their original volume of approximately 23,000 pounds of propellant, and the resultant increase in spacecraft length would not be great enough to violate the 208-inch limit.

The use of the Transtage propulsion system's two main rocket engines reduces the field of view of the medium-gain antenna so as to inhibit its use early in the interplanetary phase until the cone angle of the spacecraft-earth line decreases below 60 degrees It was decided to accept this constraint rather than the weight penalty associated with a solution such as a double gimbal and extension arm.

The modularity of the spacecraft design is very good. The propulsion subsystem can be built as a separate unit and then inserted into the spacecraft, and the various subsystems in the spacecraft are capable of being removed and installed in separate units or modules. This factor will aid in prelaunch ground handling and testing of vehicle subsystems.

No detrimental effects are expected insofar as environment imposed on the spacecraft by the propulsion system. Heat transfer effects will be limited to plume radiation and shock, vibration, and acceleration imposed by the propulsion system are well within the capability of the spacecraft structure.

Since the Transtage vehicle has been developed and qualified, this will allow use of existing ground handling equipment and procedures in preparing the Voyager vehicle for flight. Also, launch personnel will have acquired valuable experience from the 17-flight Research and Development test flight program.

4. CUSTOM LIQUID PROPULSION SYSTEM

This section presents an optional spacecraft design that utilizes the modified LEM propulsion system, but with new tankage and a new thrust mount. The propulsion system is designed as a separate module which could be inserted into the spacecraft, as are the other subsystems within the spacecraft.

The resulting design has many of the advantages of the LEM system in terms of development status and operational capability but it gains additional advantages in terms of modularity, weight, and reliability.

Its disadvantages are its length of 208 inches and its lack of adaptability for increased propellant capability.

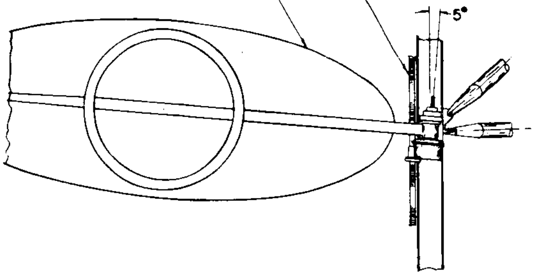
The following sections contain a description of the vehicle, additional discussion concerning the advantages of the system, and a detailed weight breakdown.

4.1 Voyager Spacecraft Design Based on Custom Liquid Propulsion Subsystem

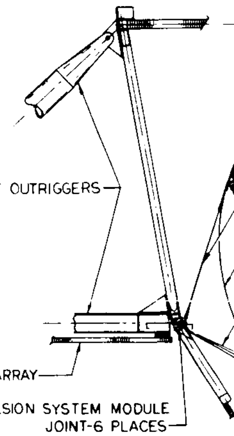
The Voyager spacecraft configuration illustrated in Figure 24 utilizes the single liquid propulsion system for midcourse correction, orbit insertion, and orbit trim maneuvers which was described in 4 of Section IV.

The propulsion module, which interfaces with the bus structure at the aft frame of the equipment compartment, consists of a modified LEM descent engine, an 82-inch-diameter propellant tank with an internal bulkhead to separate the fuel from the oxidizer, two 33-inch-diameter helium pressurization spheres and all necessary valves, lines, actuators, and regulators.

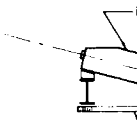
The thrust mount for the LEM descent engine is constructed of aluminum sandwich panels attached to six tubular aluminum longerons which diverge from the engine support to form a hexagon at the bus interface. The helium pressurization spheres partially intersect the two flat



VIEW **L-L**

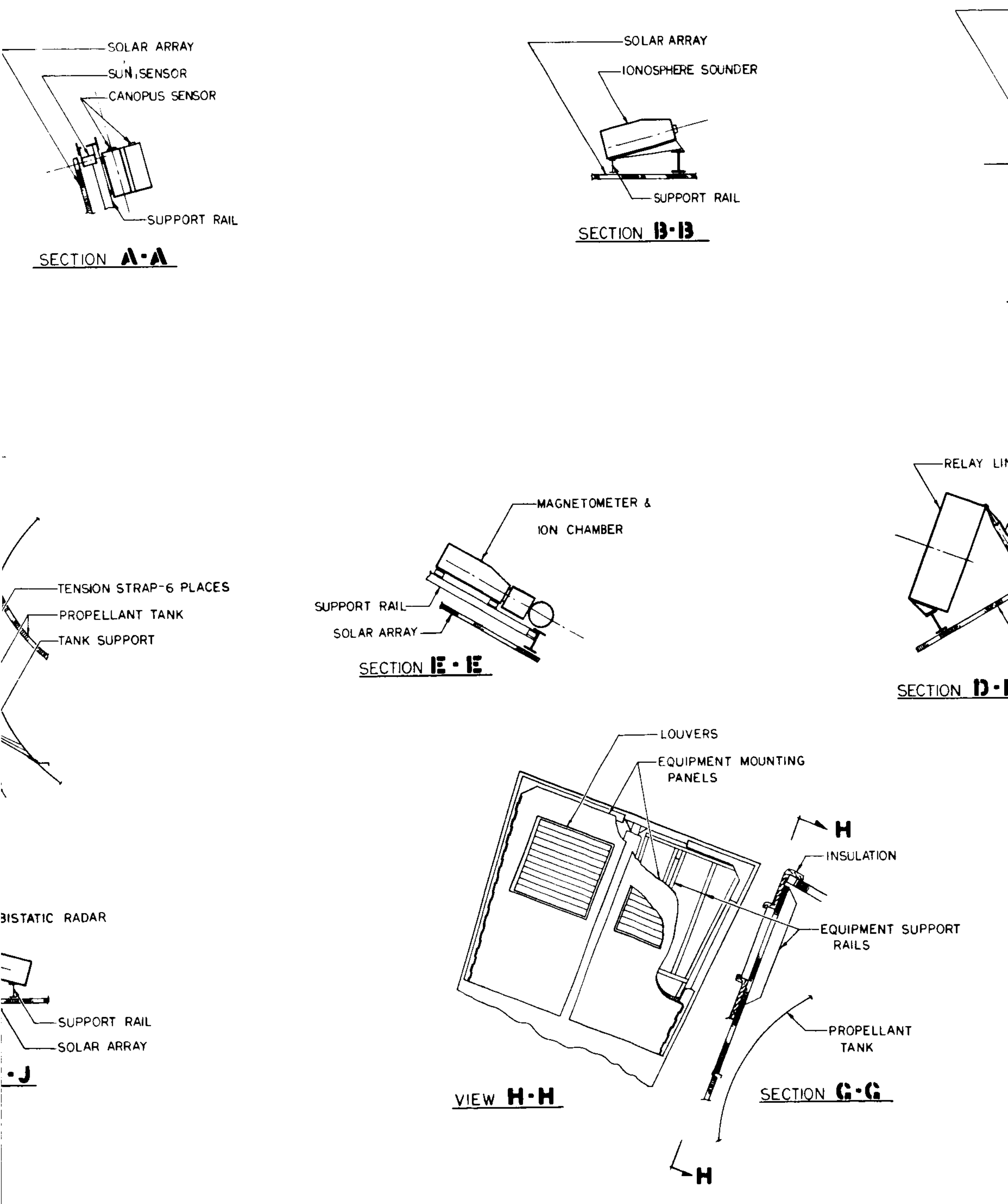


SECTION **K-K**

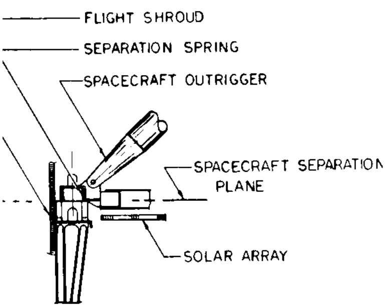


SECTION **J**

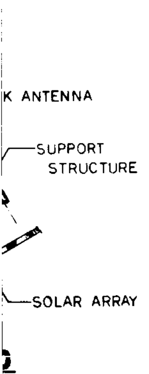
/

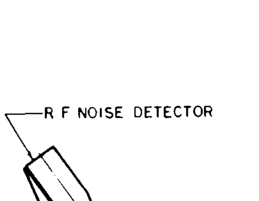


SOLAR ARRAY
SUN SENSOR
CANOPUS SENSOR
SUPPORT RAIL
SECTION A-A
SOLAR ARRAY
IONOSPHERE SOUNDER
SUPPORT RAIL
SECTION B-B
RELAY LI
SECTION D-
MAGNETOMETER &
ION CHAMBER
SUPPORT RAIL
SOLAR ARRAY
SECTION E-E
TENSION STRAP-6 PLACES
PROPELLANT TANK
TANK SUPPORT
BISTATIC RADAR
SUPPORT RAIL
SOLAR ARRAY
-J
LOUVERS
EQUIPMENT MOUNTING
PANELS
H
INSULATION
EQUIPMENT SUPPORT
RAILS
PROPELLANT
TANK
VIEW H-H
SECTION G-G
H

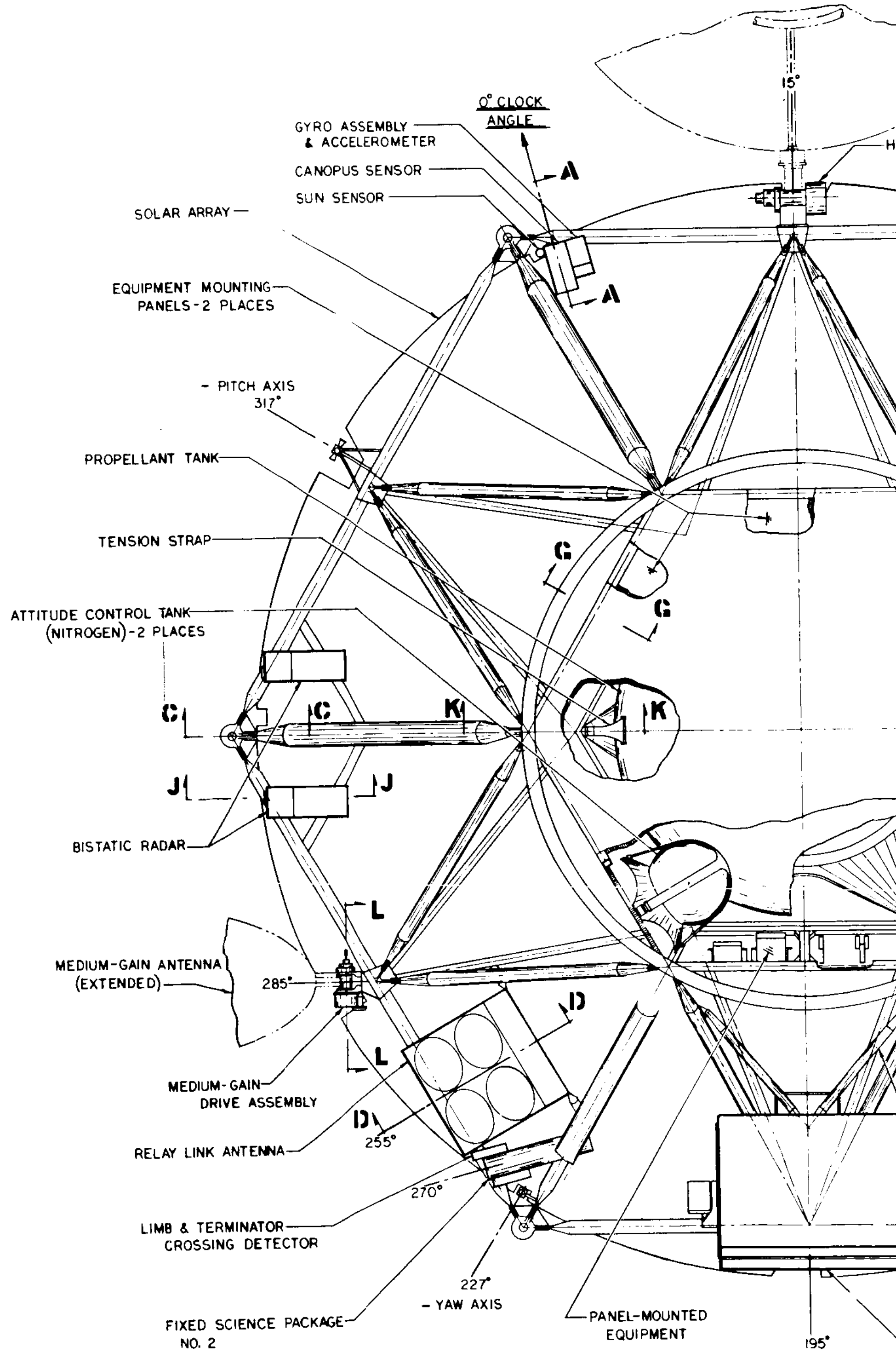


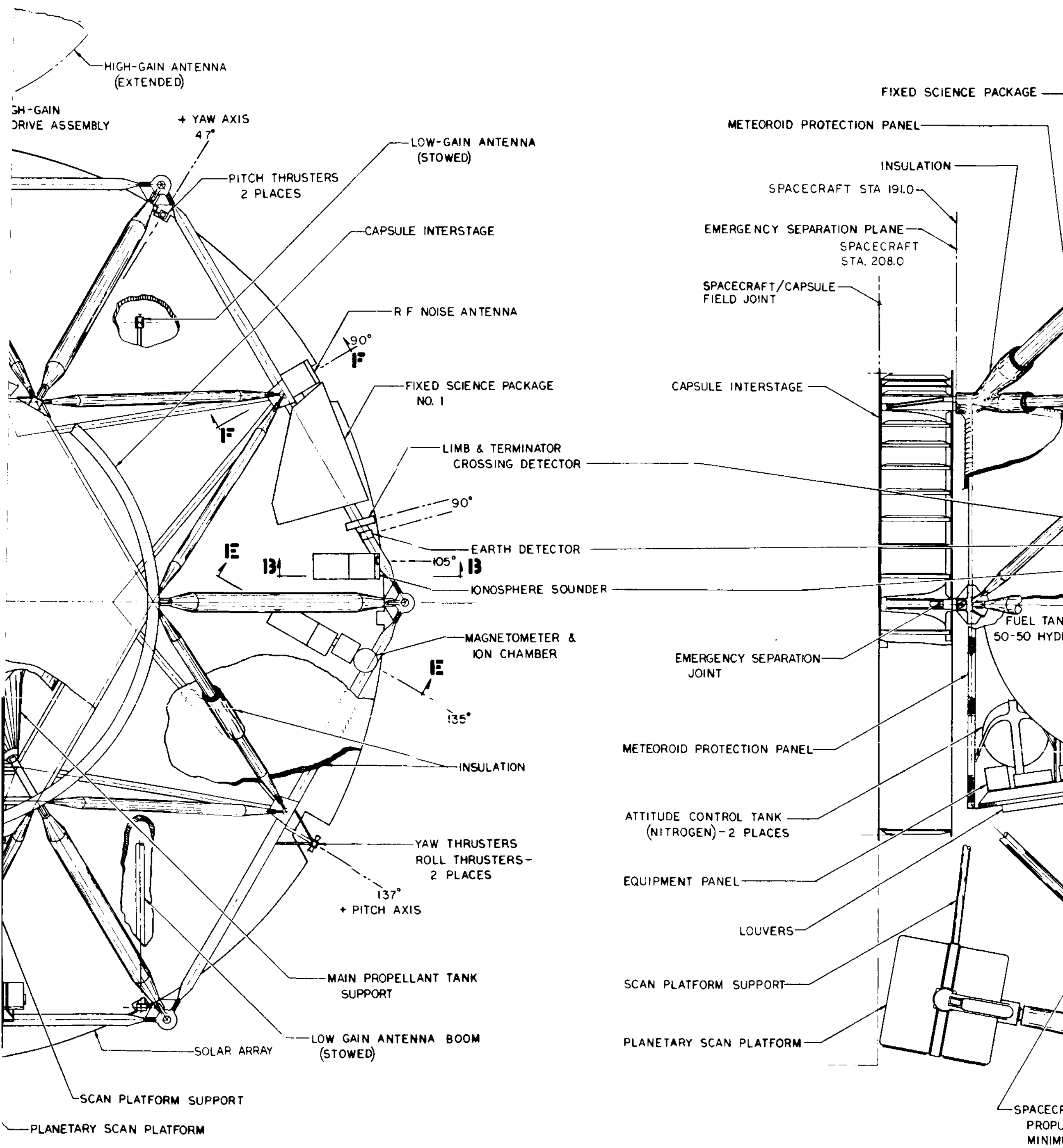
SECTION C·C





SECTION F·F




HIGH-GAIN ANTENNA (EXTENDED)
GH-GAIN DRIVE ASSEMBLY
+ YAW AXIS 47°
PITCH THRUSTERS 2 PLACES
LOW-GAIN ANTENNA (STOWED)
CAPSULE INTERSTAGE
R F NOISE ANTENNA
90°
F
FIXED SCIENCE PACKAGE NO. 1
LIMB & TERMINATOR CROSSING DETECTOR
90°
EARTH DETECTOR
E
B
105°
B
IONOSPHERE SOUNDER
MAGNETOMETER & ION CHAMBER
E
135°
INSULATION
YAW THRUSTERS
ROLL THRUSTERS - 2 PLACES
137°
+ PITCH AXIS
MAIN PROPELLANT TANK SUPPORT
LOW GAIN ANTENNA BOOM (STOWED)
SOLAR ARRAY
SCAN PLATFORM SUPPORT
PLANETARY SCAN PLATFORM
FIXED SCIENCE PACKAGE
METEOROID PROTECTION PANEL
INSULATION
SPACECRAFT STA 191.0
EMERGENCY SEPARATION PLANE
SPACECRAFT STA. 208.0
SPACECRAFT/CAPSULE FIELD JOINT
CAPSULE INTERSTAGE
FUEL TAN
50-50 HYD
EMERGENCY SEPARATION JOINT
METEOROID PROTECTION PANEL
ATTITUDE CONTROL TANK (NITROGEN) - 2 PLACES
EQUIPMENT PANEL
LOUVERS
SCAN PLATFORM SUPPORT
PLANETARY SCAN PLATFORM
SPACECR
PROPU
MINIMU

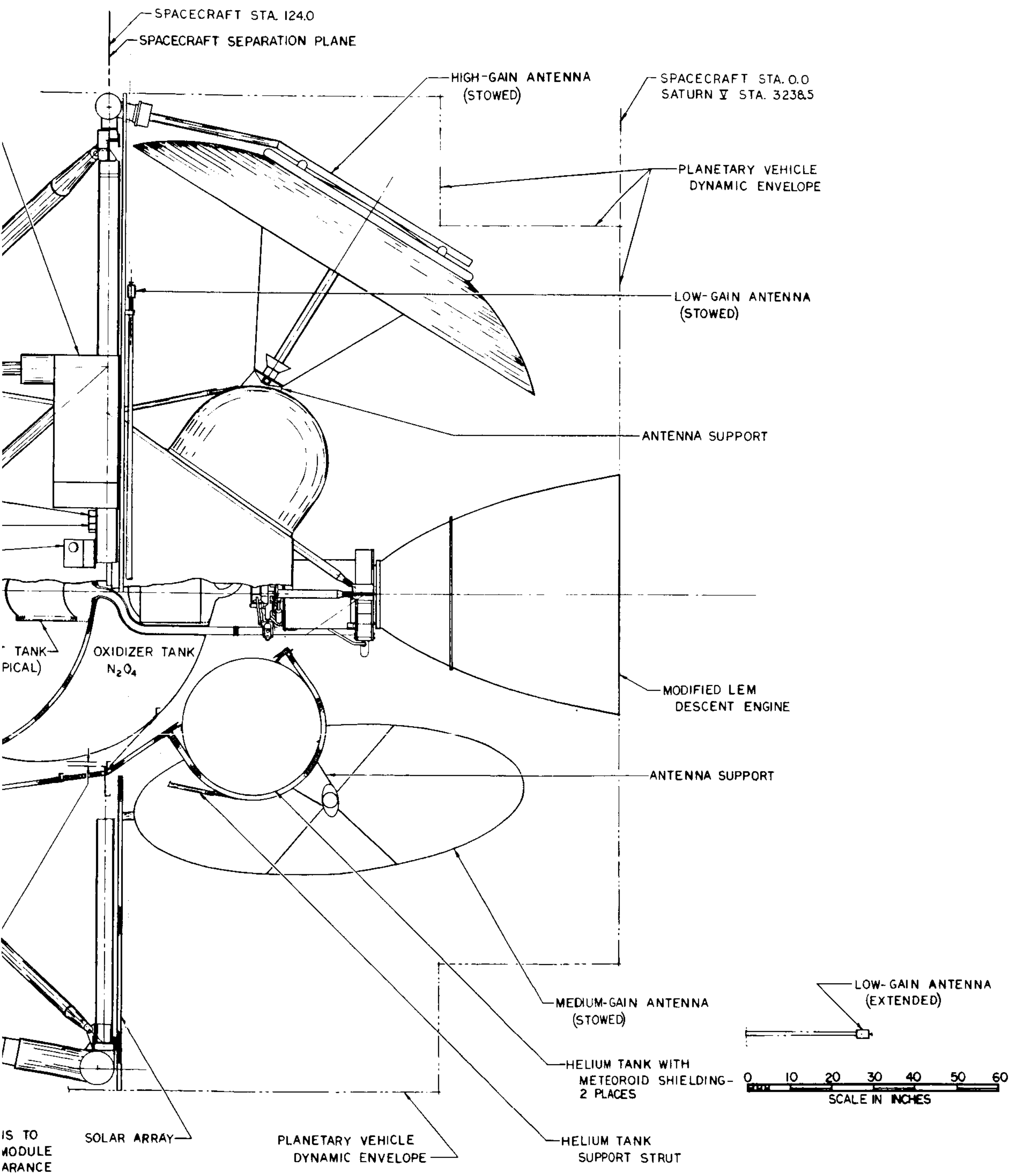


Figure 24. 1971 Voyager Spacecraft—Custom Liquid Propulsion Configuration

#5

sides of the thrust structure and are pre-loaded in place by means of a canister. An auxiliary support member for the canister is provided and interconnects with the hexagonal frame at the bus structure interface. The main propellant tank assembly is supported from the forward frame of the hexagonal thrust structure by means of an aluminum cradle which is bolted to the external flange of the tank. Lateral stability is provided by a similar cradle structure which encloses the forward end of the tank and terminates at the forward ring of the truss structure. The sandwich thrust mount and canisters serve as micrometeoroid shields and are used to support the required multi-layered insulation for thermal control. The thrust structure also provides a rigid support for the retention assemblies of the high and medium gain antennas.

The bus structure consists of the truncated equipment compartment, the fixed solar array platform, and six radial outriggers. This structure serves to react the capsule inertia load which is transmitted through the titanium semi-monocoque cylindrical adapter to the forward bulkhead of the central equipment compartment. The total planetary vehicle inertia load is then trussed into the vehicle shroud adapter. For the maximum load condition, the diagonal members carry the compressive loads into the sandwich stabilized space frame at the forward end of the equipment compartment. The horizontal truss members carry the tensile loads which are reacted in shear through the adjacent panels of the solar array. Thus, the six identical and fixed solar panels become an integral part of a rigid equipment platform which supports all appendages, the fixed science package, the reaction control nozzles, and the capsule/spacecraft antenna. Intermediate diagonal truss members, which intersect at the mid-point of the solar array outer frame, complete the space frame of the bus structure.

The geometry of the hexagonal equipment compartment was designed to meet primary structural requirements and the subsystems volume requirements, the space requirement of the 82 inch-diameter propellant tank, and the two reaction control system nitrogen spheres which are supported from the central compartment structure. Subsystem mounting requirements in addition to center of gravity and thermal control constraints required the use of four of the six bays. The face of each of the four bays is subdivided into two panels which are hinged along the outside vertical edges. These equipment or radiation panels support the sensors, the power equipment, tape recorders, science packages, command detector and decoder, and the remaining spacecraft electronic assemblies. The other faces of the compartment are also constructed of aluminum sandwich panels which are attached to the space frame of the compartment. All panels act as primary shear panels and as micrometeoroid shields for the pressure vessels and sensitive electronics.

Radiant thermal energy interchange between the main compartment and the solar array is attenuated by aluminized mylar insulation blanket. This blanket, which envelopes the exposed truss members, is installed on the back side of the solar array and is tied to the external surface of all sandwich panels. Active regulation of the radiant energy interchange between the main compartment and its environment is accomplished through a series of bi-metal actuated louvers attached to each of the equipment mounting doors. All other irregular protrusions and seams are suitably insulated to minimize leaks.

4.2 Spacecraft Configuration and Geometry Considerations

In order to establish a baseline for configuration comparison, strict adherence to three design objectives was emphasized during the evolution of the reference spacecraft. These objectives included the utilization of primary structure for micrometeoroid protection, complete modularity, and the optimization of all mechanical hardware. In addition, the propulsion module was intended to be unique so that the Transtage or LEM adaptations would not necessarily be favored.

Compliance with the allowable envelope precluded a tandem tank arrangement; a parallel tank arrangement would be similar to the alternates; therefore, a single propellant tank with a common bulkhead to separate fuel and oxidizer was selected. A spherical vessel is, of course, optimum; however, this feature is offset by the stress and loading problems associated with a common bulkhead design.

In order to provide singular load paths between the LEM engine and the capsule adapter and to adhere to the modularity objective, an aluminum sandwich thrust mount was conceived. This structure diverges pyramidally to form a hexagon at the bus interface, and is used to support the propellant tank at its forward frame and the two helium pressurization spheres within its truncated side panels. The complete propulsion module can then be installed or removed with or without the solar array in place.

The remainder of the structure is a large space frame which is completely stabilized by the four equipment mounting panels, the two auxiliary side panels, and the solar array panels. Secondary truss members are required to support the frame between the outriggers. Although the prime structural requirements have been satisfied, this concept requires that the majority of panels be installed to present a stable configuration for ground handling maneuvers. Alternate structural concepts to satisfy the later requirements would impose a severe weight penalty.

Complete modularization was compromised slightly since the solar array panels become integral with the bus compartment. Again the minimum weight target precluded the utilization of redundant aft frames which would be used to thermally isolate the array. The later approach would serve to reduce induced appendage misalignments and heater power requirements for temperature control.

Within the constraints imposed by the 240-inch-diameter shroud envelope and the 86-inch clearance diameter for the liquid propulsion module, it was determined that only 270 ft^2 of fixed solar array could be provided. Although this area does meet the minimum power system requirement of 260 ft^2, it is somewhat less than the design goal of 290 ft^2. It was also necessary to incorporate local cutouts in the solar array panels to provide clearance for the high- and medium-gain antennas which does compromise slightly the optimum cell packaging concept. Although the hexagonal form of the bus structure, with its sandwich-stabilized space frame, presents an optimum arrangement, only six points of support are available for the transmission of loads from the capsule to the forward bulkhead of the bus. Therefore, to achieve the required circumferential load distribution at the adapter field joint, the titanium semi-monocoque cylindrical adapter had to be lengthened. This change in length imposed a slight weight penalty and increased the over-all length of the planetary vehicle to the maximum allowable envelope.

The optimization of the outrigger structure takes advantage of the strength and stiffness of the solar array panels to the extent that, without these panels, the non-flight loading conditions of ground handling and test would become one of the critical design load conditions. Therefore, the ground support equipment would be complicated by the fact that it would be required to stabilize the bus structure in the event that the solar panels were removed.

The auxiliary diagonal truss members, which intersect at the mid-point of each of the six outer solar array frames and extend to the vertices of the equipment compartment structure, compromise the accessibility to the equipment compartment. One or both of these tubular members would have to be removed to gain access to the equipment mounted in the compartment. Although the removal and re-installation is undesirable, the diagonals are only required for in-flight load conditions and, therefore, their removal during the normal ground handling conditions would not adversely affect the structural integrity and the alignment of any equipment.

4.3 Weight

A sequential weight summary of the Voyager spacecraft based on the custom liquid propulsion subsystem is presented in Table 10. Also listed in this table are column totals indicating which of the weights are in the spacecraft bus, flight capsule, and propulsion subsystems. These column totals are equal to the weight allocations specified by JPL. The total weights for the spacecraft propulsion and bus are shown as specified (17,500 pounds) although the propulsion and the bus weights do not necessarily total to the 15,000 and 2,500 pounds independently. This is because of the difficulty in establishing a clearly distinguishable line between the propulsion subsystem and the spacecraft bus.

The custom liquid propulsion system is a completely new design with the only existing hardware consisting of the LEM descent engine. Since the propellant containers are sized to the exact mission requirements (12,182 pounds total usable propellant), an iteration had to be made from the detail designed configuration (12,000 pounds propellant). The latter system contains a 93.2-inch-diameter, 225 psi titanium tank with a sandwich-constructed common-bulkhead, and four 26-inch-diameter,

Table 10. Voyager Planetary Vehicle Weight Summary (Custom Liquid Propulsion Configuration)

Item	Capsule Weight	Propulsion Weight	Bus Weight	Total Weight
Spacecraft Bus				
Structural and mechanical		692	656	1,348
Pyrotechnics			51	51
Temperature control		49	114	163
Radio			126	126
Relay link			25	25
Data storage			72	72
Telemetry			8	8
Command			11	11
Computing and sequencing			36	36
Cabling			229	229
Power			522	522
Guidance and control			268	268
Balance weights			15	15
Contingency			135	135
Spacecraft Propulsion				
Propulsion inert weight		1,627		1,627
Start system inert weight		35		35
Interplanetary trajectory correction inert weight*				
Contingency		98		98
Unseparated Capsule Interstage, etc.	250		149	399
Spacecraft Science Payload and Support			400	400
Flight Spacecraft Burnout Weight	250	2,501	2,817	5,568
Flight capsule	2,490			2,490
Jettisoned canister	260			260
Orbit trim propellant (100 meters/sec)		300		300
Planetary Vehicle in Orbit	3,000	2,801	2,817	8,618
Propellant for Mars orbit insertion		10,482		10,482
Inerts expended				
Planetary Vehicle After Interplanetary Trajectory Correction	3,000	13,283	2,817	19,100
Interplanetary trajectory correction propellant (200 meters/sec)		1,400		1,400
Planetary Vehicle Gross	3,000	14,683	2,817	20,500
Planetary Vehicle Adapter				1,500
Total Weight	3,000	17,500		22,000

*Propulsion subsystem serves this function also.

3000 psi titanium pressure bottles. Propellant settling is obtained by utilizing the two start tanks described in the LEM configuration discussion.

The following subsystem weights are assumed to be constant for all configurations and are discussed in Volume 2:

Radio	Command
Relay Link	Computing and Sequencing
Data Storage	Cabling
Telemetry	Power

The spacecraft structural and mechanical subsystems are essentially of the same type construction as the LEM configuration. However, the custom design utilizes the lower member of the outriggers as an integral part of the solar array support structure, and truss structure is added to provide further support. Also, the solar array linkage system has been deleted.

4.4 Environment Imposed on the Spacecraft and Capsule

The environment imposed on the spacecraft by the custom liquid propulsion system will be similar to that previously reported for the modified LEM descent stage. Solar cell thermal requirements dictated replacement of radiation skirt with an ablative skirt to reduce radiation from the nozzle. As shown in Appendix C radiation from the exhaust plume will be restricted to less than 10 Btu/hr-ft^2 at all points of the solar array. Total array temperature rise due to these two sources will be less than 5^{o}F.

Acceleration levels for this propulsion system can be held to a maximum of 1 g which will occur at the end of the orbit insertion maneuver. The higher thrust level of the LEM descent engine for this configuration can be adjusted to the value of 6400 pounds thrust required to accomplish this. Vibration inputs of 15 to 2000 cps are expected from the propulsion with an acceleration level of 1.92 g's in the 20 to 100 cps range.

No detrimental effects on the spacecraft or limitations thereon were encountered in the design due to these propulsion system imposed environments.

4.5 Other Considerations

The reliability analysis of the custom liquid configuration differs from that of the LEMDS configuration only slightly--the number of propellant tanks has been reduced from four to two. The assessment of Appendix A gives a probability of success of the propulsion system of 0.969, and no degradation of the spacecraft bus reliability was considered to be imposed by the propulsion system.

The comparative costs associated with the spacecraft bus for this configuration are modest. They are due to bus structure and mechanical subsystems, and amount to $8.5 million for development, and $10.7 million for production costs for the 1971 mission.

4.6 Summary

The weight of the custom liquid propulsion system is 14,683 pounds, of which 12,182 pounds is usable propellant, resulting in a mass fraction of 0.83. This high mass fraction resulted from a more compact arrangement of the propulsion system that had a minimum effect on configuration, geometry, and look angles of the various subsystems within the spacecraft.

The length of the vehicle is 208 inches, the maximum allowed by the specification. Any future additional propellant requirements would cause an increase in this dimension unless major revisions were undertaken in the spacecraft design.

The modularity of the spacecraft design is very good. The propulsion system is inserted as a separate unit and bolted in place, and all other equipment and panels can be removed separately.

Since this propulsion system utilizes the LEM engine, the identical advantages accrue in the areas of prelaunch ground handling techniques and operational support equipment that were previously described in the modified LEM system.

VI. SYSTEM COMPARISON

1. APPROACH USED FOR PROPULSION SYSTEM COMPARISON AND SELECTION

The alternate propulsion systems studied in this volume must be compared and evaluated from a standpoint of meeting the design, functional and performance criteria established for the Voyager system. The requirements of the 1971 mission and subsequent missions must be taken into consideration. This is done by the presentation of Section III and by the analyses of Sections IV and V, and Appendixes A and B.

The approach followed is a qualitative one, in which the alternate spacecraft-propulsion configurations are compared with each other for each of the various criteria, and judgement is exercised to attach the appropriate relative importance to each comparison and to select the best choice.* The results, in the present instance lead to a clear-cut identification of the superior propulsion system alternate - the LEM descent stage configuration.

2. CRITERIA FOR COMPARISON

Major criteria for comparison were derived in Section III, and discussed in relation to the JPL-defined five competing spacecraft design characteristics. These comparison criteria include (ranked in the order of relative importance):

- Probability of success
- Performance
- Cost
- Flexibility
- Effects on spacecraft design

*The qualitative approach, rather than a quantitative one, was chosen because of the difficulty in devising a meaningful numerical rating scheme. This difficulty is discussed in 3 of Section III.

- Compatibility with planetary quarantine requirements
- Compatibility with prelaunch ground handling sequence
- Modularity
- Testing and MOSE requirements

Of course, each of these criteria is comprised of a number of facets, as outlined in Section III, and all of these facets are considered in the comparison.

3. QUALITATIVE COMPARISON OF FOUR CANDIDATE PROPULSION SYSTEMS

Table 11 compares the three principal candidate propulsion systems, i. e., the modified Minuteman second stage with liquid monopropellant midcourse system, modified LEM descent stage, and modified Transtage; plus the custom-designed liquid propulsion system. Categories of comparison are those listed in Section 2. Asterisks indicate features in each candidate configuration which present a major relative advantage.

The salient points of the comparison may be summarized as follows:

3.1 Probability of Success

The numerical probability of mission success abstracted from Appendix A accounts for component reliability as reflected in published failure rate data, design simplicity, and component redundancy. The superiority of the LEMDE-based alternates - the LEMDS configuration and the custom liquid configuration - stems principally from the design simplicity in the use of a single engine and a single pressurization-propellant feed system for all the propulsive requirements of the mission.

Not incorporated in these assessments are specific areas noted (effect of the solid-engine environment; uncertainty of factors leading to stress corrosion for liquid systems), developmental maturity of the design, and functional redundancy or flexibility leading to failure-mode operation or partial success.

Table 11. Comparison of Four Candidate Propulsion Systems

COMPARISON FACTORS	SOLID	LIQUID SYSTEMS		
	A Minuteman Wing VI Stage 2 (modified for Voyager) plus monopropellant midcourse	B LEM Descent Stage (modified for Voyager)	C Titan III-C Transtage (modified for Voyager)	D Custom Liquid Propulsion System
1. PROBABILITY OF SUCCESS				
Assessed value for sample mission profile (Appendix A)	0.949	0.968	0.924	* 0.969
Principal areas of uncertainty	Effects on spacecraft due to engine exhaust plume	Possible degradation in reliability due to stress corrosion of titanium propellant tanks by N_2O_4. (Minimized by reducing tank pressure during interplanetary phase)		
Developmental maturity	*Considerable flight experience; substantial modifications	Flight experience late '60s; *minimum modifications	Considerable flight experience; substantial modifications	New tankage development LEM engine
2. PERFORMANCE OF 1971 MISSION				
ΔV for orbit insertion, km/sec (based on allocated weights)	2.00 (Satisfies requirement)	2.10 (Satisfies requirement)	2.29 (Exceed desired value)	* 2.37 (Exceed desired value)
Minimum ΔV ΔV error	OK for midcourse and orbit trim Highest error for orbit insertion	*OK	OK, but jeopardized by limited propellant for auxiliary engines	*OK
3. COST ($ MILLIONS)				
Propulsion system and bus structure and mechanical subsystems (Appendix B)				
Development	47.7	* 28.1	40.3	52.9
Production — 1971 mission	30.8	27.1	26.3	26.7
Total	78.5	55.2	66.6	79.6
4. FLEXIBILITY				
Propellant sources for high and low thrust	Separate	* Common	Separate	* Common
Variable ΔV for orbit insertion and accommodating mass change	No	* Yes	* Yes	* Yes
Orbit insertion ΔV for 1975-77 weight allocations, km/sec	1.11 (Sub-marginal; may be increased 5% by using Beryllium propellant)	1.20 (Acceptable)	1.30	1.35
Ability to produce greater impulse for future missions	Requires new solid motor development	*Excess propellant capacity	Excess capacity if Transtage tanks restored	Requires new design
5. EFFECTS ON SPACECRAFT DESIGN				
Flight spacecraft length	208 in.	208 in.	192 in.	208 in.
Cross section area for power	*Fixed array	*Fixed array	Deployable panels required for some solar array area	*Fixed array
Required by propulsion environment	Deployable heat shield to protect solar cells Protection for PSP Low-gain antenna abandoned or stowed	Ablative nozzle extension	Ablative nozzle extension	Ablative nozzle extension
6. HAZARD TO PLANETARY QUARANTINE	Possible ejection of contaminated solid particles after burnout			
	Possibility of meteoroid-induced rupture of propellant tanks leading to structural disintegration and ejection (Minimized by lower cross section of monopropellant tanks)	(Minimized by reducing tank pressure during cruise)		
OUTSTANDING ADVANTAGES	• Flight experience • Simplest main engine	• Probability of success • Lowest cost • Flexibility		• Probability of success • Performance
OUTSTANDING DISADVANTAGES	• Exhaust plume problem • Inflexibility • Cost of development		• Scope of modifications • Probability of success	• Cost of development • Development status

* Indicates superiority

3.2 Performance of 1971 Mission

All candidate systems are satisfactory from a standpoint of achieving the ΔV performance required by the mission. Payload capability increases from solid plus monopropellant system, to LEMDS to Transtage and to customized configurations. Mission flexibility and emergency backup mode capabilities offered by the liquid systems are distinct advantages over the solid/monopropellant system.

3.3 Cost

Development cost considerations favor the LEM descent stage system in view of its minimal modification needs, compared to the cost of the extensive redesign for Transtage and the solid/monopropellant system and for the custom-designed liquid system which would have to be developed. Thus early outlays for developmental programs are minimized. The production costs of all systems are similar, so the LEMDS configuration is lowest in total cost.

3.4 Flexibility

This category includes additional 1971 mission capability and contributions to future missions (as indicated in the Voyager tentative mission plan or otherwise) as outlined in 2.4 of Section III. Generally, the liquid systems are superior to the solid motor configuration in this respect. In particular, the LEMDS configuration, with its common propellant supply for high and low thrust levels and its excess propellant capacity, exhibits the greatest mission flexibility. Its orbit insertion ΔV for 1975-77 weight allocations, while greater than that of the solid motor, is the lowest of the liquids. However, it is easy to see how this capability can be enhanced - by use of a nominal 8000-pound capsule rather than 10,000 pounds, or by devoting some of the additional 1000-pound bus allocation for 1975-77 to increased propellant.

3.5 Effects on Spacecraft Design

This section summarizes points developed in Sections IV and V. To a large extent, they have already been accounted for in the reliability comparison (if they complicate the design on the mission sequence) or in the performance comparison (if they impose weight penalties). Thus, this comparison per se is not given great importance in the selection.

3.6 Hazard to Planetary Quarantine

This criterion, while of primary importance to the Voyager mission, is difficult to evaluate with respect to propulsion systems. (Reference is made to Volume 1, Appendix E, which considers possible contamination processes based on liquid engines.) Offsetting factors seem to make solid and liquid propulsion systems about equally desirable from this point of view. The LEMDS configuration presents less hazard than the other liquid systems, and possibly the least of all alternates.

3.7 Other Considerations

Other criteria include compatibility with the prelaunch ground handling sequence, modularity, and testing and MOSE requirements. While these are important factors to the conduct of the program, they were not listed in Table 11 because there were not significant differences in the implications of these criteria on the alternate configurations. To the extent the effects of these criteria can be measured in dollars, they are treated (but considered essentially equal) in Appendix B.

The results of the qualitative comparison are summarized at the bottom of Table 11. The configuration based on the LEM descent propulsion stage is selected because it

- Has superior probability of success
- Has adequate performance for the 1971 mission
- Requires the least cost
- Requires the least modifications
- Has the greatest flexibility for additional 1971 mission capability as well as application to future missions

4. TASK B VERSUS TASK A

In the Phase IA (Task A) Study TRW selected a solid motor (with monopropellant liquid midcourse engine) for the Voyager spacecraft propulsion system. Here in Task B we have selected a liquid engine - the LEM descent propulsion stage. It is certainly pertinent to examine and review the study constraints which led to the reversal in recommendation.

First, the increased size of the propulsion system, with the required available impulse about four times as great in Task B as in Task A, has made it both possible and desirable to examine the applicability of available current propulsion system developments. In Task A, choice of either liquid or solid propulsion entailed essentially a complete development. Whereas current solid engines seem to be sized as close to the Task B requirements as liquid engines are, relatively small changes in requirements result in a more substantial development program for solid motors.

Second, the adverse heating of the spacecraft by the exhaust plume of the solid motor is more serious in Task B. In Task A, a solid motor was acceptable in this regard because (1) the smaller total impulse resulted in a much lower total heat flux, and (2) the mission profile called for use of the engine after the capsule was jettisoned, so that the plume could be directed opposite to the direction faced by the solar cells. Thus, in Task B, a much larger heat flux would impinge on the solar array side of the spacecraft. In terms of heat absorption (and consequent temperature rise), the Task B situation represents an aggravation of the problem by a factor of 20: the integrated heat flux impinging on the surface is four times as large, and the solar cells have an absorptivity five times as great as the reflective coating which would be applied to the opposite side of the spacecraft as in Task A. In addition, the solar cells are more sensitive to increased temperature than other exterior components. Although an engineering solution was found to protect the solar array, this solution carries a weight and reliability penalty, and does not resolve all the problems associated with exhaust plume heating.

Third, the adoption in Task B of the descent-from-orbit mode for the lander has put a premium on the flexibility of orbit attainment by the spacecraft. Evidence of this recognition of the versatility desired lies in JPL study input data:

- Voyager 1971 Preliminary Mission Description, October 15, 1965, page 36, describes the extent of apsidal rotation desired in the establishment of the orbit.
- The addendum of November 22, 1965 on the capsule-spacecraft communications requirements imposes severe constraints on the orbit to be achieved.

Neither of these statements prohibits the use of a solid motor with its fixed impulse for orbit insertion; however, both indicate the desirability of flexibly controlling the orbit insertion in a way which is much simpler with the variable impulse of a liquid system.

Finally, in Task B, JPL for the first time has explicitly listed cost as a competing characteristic to be considered.

All of these changes in study constraints have contributed to the Task B TRW recommendation.

APPENDIX A

PROBABILITY OF SUCCESS

1. INTRODUCTION

The purpose of this appendix is to generate and present data giving comparable probabilities of successful operation of the alternate propulsion subsystem-spacecraft system configurations studied in this volume. For a representative mission, the reliabilities of the alternate propulsion systems are assessed, and, where appropriate, the reliabilities of those spacecraft system components (outside the propulsion subsystem) which are peculiar to the choice of propulsion subsystem are assessed.

Because these alternates comprise engines already developed for other programs as well as components and systems which would be developed for Voyager, the reliability source data are necessarily diverse as to source, vintage, and appropriate interpretation. It is intended that the comparisons drawn here validly account for this diversity.

2. BASIS

A single sample mission profile was generated for the determination of probability of success of all alternates. It consists of launch, a 6-month interplanetary cruise during which three midcourse corrections are interspersed, insertion into orbit about Mars, and one orbit trim maneuver conducted after 50 hours in orbit. While this mission profile does not represent the maximum mission demand in terms of lifetime and number of engine operations, it is representative of typical Voyager 1971 missions in life and complexity. It is summarized in Table A-1. The differences in the columns for the alternate systems are only those resulting from different engine thrust levels, leading to somewhat different engine operating times for the same mission. (These differences have a minor effect, because, as indicated by the analysis, the 6-month dormant period in transit represents the dominant degrading influence on subsystem success for the mission.) The sample mission makes no allowance for the value of a partially successful mission, which could occur even though not all the propulsion functions are performed.

Table A-1. Mission Profile for Reliability Analysis

Phase	Combination Solid plus Liquid Propulsion	LEM Descent Propulsion Stage	Transtage Propulsion	Custom Liquid Propulsion
	Mission Time, hr			
Boost	0.3	0.3	0.3	0.3
Interplanetary Cruise	4320.	4320.	4320.	4320.
Midcourse Corrections (interspersed during interplanetary cruise)	0.111[b]	0.111[b]	(c)	0.111[b]
Orbit Insertion	0.018[a]	0.089[a]	(c)	0.089[a]
Orbit Cruise	50.	50.	50.	50.
Orbit Trim	0.027[b]	0.055[b]	(c)	0.055[b]

Sources:
(a) Primary, or high-thrust engine firing
(b) Secondary, or low-thrust engine firing
(c) These propulsion operations will involve both main and auxiliary (attitude control) engine firings, allocated as 0.065 hour of main engine operation and 0.055 hour of auxiliary engine operation

The extent or domain of the reliability analysis is also intended to be comparable for the alternate systems analyzed. It is essentially a comparison of all reliability effects attributable to the propulsion subsystem; but this includes not only the propulsion elements themselves, but components of the spacecraft which are peculiar to the particular propulsion design. In this latter category are included:

- In the combination solid-liquid propulsion configuration, the operation of the solar-panel shield, necessary to avoid the extreme environment imposed by the engine exhaust plume

- In the Transtage configuration, the release and deployment of four small solar array panels, which is necessitated by the fact that the 10-foot diameter of the Transtage propulsion module occupies too much of the available projected cross section area for a fixed solar array to meet power requirements.

The domain of the analysis is limited in other dimensions by these interface definitions:

- The mechanical or pneumatic means of accomplishing thrust vector control during engine operation is included in the analysis.

- The commands and electrical means of actuating thrust vector controls are excluded.

- All attitude control functions when the engines are not firing are excluded.

- Structural reliability is generally excluded. In this regard, it is noted that within the spacecraft and propulsion weight summaries given in this volume is included provision for meteoroid protection leading to approximately the same probability that no meteoroid penetrations will occur in propellant tanks or solid motor (.988 in 6 months) for each alternate design. To the extent that the different alternates require different structural weights to achieve equal structural reliability, this is accounted for in the calculation of ΔV performance capability in this volume.

3. RESULTS

The assessment of the probability of successful operation is made for these five configurations:

- Combination solid-liquid propulsion configuration (monopropellant midcourse engine)
- Alternate combination solid-liquid propulsion configuration (bipropellant midcourse engine)
- LEM descent propulsion stage configuration
- Transtage configuration
- Custom liquid propulsion configuration.

These are the major alternates discussed in Volume 5.

The results of the reliability analysis are given in Table A-2. For each alternate configuration there is given an "over-all probability of success" which applies to the operation of the propulsion system domain discussed above over the duration of the sample mission. This probability is resolved by a coarse breakdown in Table A-2, and supported by the detailed analysis presented in succeeding tables of the appendix. Table A-3 is a guide to this detailed analysis. Briefly, it

- Identifies the configuration by reference to sections of Volume 5, and to figures in this appendix, and defines the mission profile for that configuration by reference to Table A-1.
- Refers to the basic applicable component failure rates
- Defines the applicable environmental K factors which modify failure rates for different mission phases
- Locates the detailed analyses for each configuration.

(The tables and figures referred to in Table A-3 are all at the end of this appendix.)

In addition to the "over-all probability of success" given, Table A-2 lists an "adjusted probability of success" in which factors are introduced for which quantitative supporting data are not available, but which are felt to have a real influence on mission success. These factors are discussed in Paragraph 5 below.

Table A-2. Probability of Success (Summary)

	Propulsion System Alternate									
	Combination Configuration			LEM Descent Stage Configuration Component Failure Rates, per			Transtage Configuration		Custom Liquid Configuration (FARADA, TRW)	
		Solid plus Monopropellant Liquid	Solid plus Bipropellant Liquid		FARADA, TRW	Grumman-Apollo				
Primary propulsion system	Solid motor igniter	.9978	.9978	Propulsion system	.9736	.9917	Two main engines	.9704	Propulsion system	.9744
	Solid motor	.9814	.9814							
Primary engine thrust vector control	LITVC	.9950	.9950	Gimbals	.9940	.9996	Gimbals	.9916	Gimbals	.9940
Total primary propulsion system		.9743	.9743		.9678	.9913		.9622		.9686
Auxiliary propulsion system and associated thrust vector control		.9946	.9804					.9608		
Reliability of special spacecraft components required by propulsion system	Heat shield deploy and restow	.9993	.9993				Solar panel release	.9990		
Overall probability of success		.9684	.9545		.9678	.9913		.9236		.9686
Adjusted probability of success		.949	.935		.968			.924		.969

Table A-3. Guide to Probability-of-Success Calculations

	Propulsion System Alternate					
	Combination Configuration		LEM Descent Stage Configuration — Component Failure Rates per		Transtage Configuration	Custom Liquid Configuration
	Solid plus Monopropellant Liquid	Solid plus Bipropellant Liquid	FARADA, TRW	Grumman-Apollo		
Descriptions of configurations[2]	IV.1, V.1	(3)	IV.2, V.2	IV.2, V.2	IV.3, V.3	IV.4, V.4
Propulsion schematic diagrams	Fig. A-1, A-2	Fig. A-1, A-3	Fig. A-4[4]	Fig. A-4	Fig. A-5, A-6	(5)
Mission profile, Table A-1:	Col. 1	Col. 1	Col. 2	Col. 2	Col. 3	Col. 4
Summary component failure rates, Table A-4:	Col. 1	Col. 1	Col. 1	Col. 3[1]	Col. 2	Col. 1
Environmental K factors	Table A-5	Table A-5	Table A-5	Table A-7	Table A-6	Table A-5
Detailed calculation of probability of success	Tables A-8, A-9, A-10	Tables A-8, A-9, A-11	Table A-12	-	Table A-13	-

[1] These component failure rates are inferred primarily from Grumman reliability data presented in the JPL memorandum dated 12 November 1965: Design Data for Candidate Voyager Spacecraft Propulsion Systems, page 74, based on an Apollo mission profile and environmental K factors of TRW Report 843-6145-SC000. These environmental K factors are those of Table A-7. The Grumman component reliability data are repeated in this appendix in Table A-14.

[2] References are to sections of the body of Volume 5.

[3] The bipropellant midcourse engine was dropped in favor of monopropellant for the preferred combination liquid-solid configuration.

[4] Reliability analysis for LEMDS configuration is erroneously based on separate start tanks. This analysis is slightly pessimistic in comparison with the proposed system (Volume 5, Section IV.2) which employs start tanks internal to the propellant tanks.

[5] Custom liquid configuration schematic diagram not given. It is the same as the LEMDS configuration, except that it has only one fuel tank and one oxidizer tank.

The resulting probabilities of success, abstracted from Table A-2 are:

Configuration		Over-all Probability	Adjusted Probability
Combination solid-monopropellant		.9684	.949
Combination solid-bipropellant		.9545	.935
LEM descent stage	TRW	.9678	.968
	Apollo	.9913	
Transtage		.9236	.924
Custom liquid propulsion		.9686	.969

The adjusted probabilities of success are highest for the alternates based on the LEM descent engine—the LEM descent stage and custom liquid configurations—next highest for the solid-motor configurations, and lowest for the Transtage.

4. ANALYTICAL METHODS

For components whose performance is a single event essentially independent of storage time preceding the event, the reliability, R, is merely the probability of a successful performance of the event, as established by type approval testing. Such components include solid motor igniters, squib-actuated valves, and other electroexplosive devices. Most of the components of the propulsion system, however, must perform over a finite time span, and the probability of successful performance decreases with the duration of the event, and with the length of time of exposure to all mission operating conditions preceding the event. The rate of degradation of the probability of successful operation is greater during phases of severe environmental stress than during more benign periods; this is accounted for in the mathematics of reliability by applying a single failure rate, λ (that corresponding a quiescent or benign environment), to a component, and accounting for the increased degradation in severe environments by multiplying the actual time of exposure by an environmental factor (K factor) to give a higher equivalent time. Thus the contribution of a single phase (i) to the reliability of a component is

$$R_i = e^{-\lambda K_i t_i}$$

The probability of n identical components all surviving the phase unimpaired is

$$R_i = e^{-n\lambda K_i t_i}$$

and the probability of surviving all phases (e.g., launch, cruise, etc.) is:

$$R = \prod_i R_i = \prod_i e^{-n\lambda K_i t_i} = e^{-n\lambda \Sigma K_i t_i}$$

The term $\Sigma K_i t_i$ is the equivalent time of the mission. It may be different for different components. In this analysis, the above equation is used where the n identical components are nonredundant. Where redundancy is effected in the use of components, the calculation of R accounts for this redundancy appropriately.

Three general sources of component failure rate data were used in the analysis: that from FARADA* and TRW in-house experience; the Martin Company (for Transtage); and Grumman via JPL (for LEM descent stage, based on the Apollo mission). Examination of comparative failure rates in Table A-4 shows wide variations between these sources. In particular, the Grumman data is optimistic by about two orders of magnitude, compared with reliability estimates from the other two sources.

The imposition of environmental K factors also varies widely between the analytical methods commonly associated with the use of the different data sources. These differences are indicated by Tables A-5, A-6, A-7. In this instance, however, the Grumman-Apollo analysis employs more severe environmental factors than the other analyses do, thereby partially offsetting the effect of the optimistic component reliabilities. A different format for defining the K factors precludes direct comparison; however, for all three methods, the interplanetary cruise time has the greatest influence on the reliability assessment. For this phase, the TRW and Martin techniques both use a K factor of 1 (although

*Bureau of Naval Weapons Failure Rate Data Handbook a "criticality" factor of 0.5 is also applied, and Martin uses a modifier of 0.1 for certain "nonoperating" components), whereas the Grumman-Apollo approach uses the higher K factor of 20 for all pressurized components during this phase.

Thus the Grumman-Apollo method, using failure rates about 0.01 times TRW's, and environmental factors 20 to 40 times as great, results in probabilities of mission failure one-fifth to two-fifths those of TRW. In fact, when applied to the LEM descent stage configuration for the sample mission, the two approaches give probabilities of failure in the ratio (1 - .9913) : (1 - .9678) = .0087: .0322 = .27. In order to arrive at realistic, comparable probabilities of success we have, as indicated in Table A-2, retained the results of only the TRW approach for the LEM descent stage configuration. This is not to imply that the Grumman-Apollo approach is unrealistic. It would not be unreasonable for a program with the very extensive verification testing of flight hardware which is justifiable for the Apollo project to actually result in effective component reliability which is substantially greater than that which will be achieved for an unmanned program.

With regard to the comparison of TRW and Martin data and analyses, the two approaches appear to be about equally conservative, but the flexibility of the Martin K-factor allocations permits the results to depend strongly on the analysts' interpretations. Comparison of columns 1 and 2 in Table A-4 indicate greater reliability for LEM components than the corresponding Transtage components by a factor of 2 or 3; this is presumed to reflect truly the different demands of the Apollo and Air Force missions. We feel that the Martin analytical method might, on the average, be slightly more pessimistic about probability of success of a system than the TRW assessment of the same system. However, rather than conduct a separate TRW analysis for Transtage, and disregard the results of a study within Martin ground rules (as in the case of the Grumman-Apollo approach), we have conducted the analysis within the format of the Martin approach, but have used more optimistic interpretations where permitted by the flexibility of the method, to insure comparability of the results with the TRW analyses of the other alternates. Specifically, where Martin component failure rate data were lacking (e.g., where a modification is introduced) care was taken to not only use failure rate data compatible with TRW inputs, but to see that the environmental factors were interpreted so that this component had the same influence on the probability of success of the Transtage configuration as it would have if in the LEM descent stage configuration.

For the custom liquid configuration, no separate analysis was required. The only difference, from the LEM descent stage configuration, is the reduction in number of propellant tanks from four to two. The probability of success was merely raised .0008 to account for this change.

5. ADJUSTMENTS OF THE PROBABILITY OF SUCCESS

The calculations of probabilities of mission success based on published failure rate data do not tell the whole story. For some of the components which will comprise the propulsion subsystem (or indeed, the entire Voyager spacecraft system) no published test data are presently available, because these components have not been developed far enough. For almost all of the components we are concerned with there are no test data encompassing the entire scope of environments which will be encountered during the Voyager mission. The detailed analyses of this appendix attempt to bridge these gaps by using test data of similar components, where the actual component is as yet untested, and extrapolating existing tests to the Voyager environment by the use of K factors described above. Yet, where engineering judgment indicates that the use of past test data falls short of a realistic prediction of the probability of success, we have indicated this by adjusting the calculated probability. The principal objective of this process has been to improve the comparison of the different alternates; therefore, more emphasis is placed on relative reliability than on absolute reliability.

With regard to the probabilities calculated for successful operation of the solid motor, it was felt that the values of .9814 and .9950 for the motor itself and the liquid injection thrust vector control should be decreased, primarily because the "single event" handling of these items in the analysis does not account for their susceptibility to reliability degradation during the 6-month cruise phase. We felt the figures should be reduced to .976 and .985, respectively, to properly represent this degradation.

On the other hand the solid motor failure rate itself may well be too low. The indicated figure is supported by published test data; however, tests for which data are not yet published have come to our attention, showing somewhat lower failure rates. For this reason the solid motor reliability number is raised by .010, so that motor and LITVC are adjusted to .986 and .985, respectively.

Both the deployment of the heat shield to protect the solar cell array from excessive heat radiated from the exhaust plume during solid motor firing, and its refolding after firing to again expose the array to the sun are critical to the success of the mission. We felt that the analytically derived reliability, .9993, while a correct representation of the reliability of the components described—mostly redundant, interior devices protected from vacuum and heat—fails to reflect the degrading influences of 6 months exposure of the shield petals and hinges to the vacuum environment, and the exposure of these components to the equivalent heat of some 20 suns (actually 40 suns at 1.4 AU distance) while the motor is firing. This feeling was reinforced by comparing with .9990 for the much simpler Transtage solar panel release, a true one-shot event soon after injection. The result was to adjust the heat shield reliability figure from .9993 to .985.

The above adjustments apply equally to the solid-monopropellant and solid-bipropellant configurations.

No corresponding adjustments were deemed necessary to the probabilities of success for any of the liquid-engine alternates.

6. PRINCIPAL AREAS OF UNCERTAINTY

It is appropriate to discuss those facets of operation of the Voyager spacecraft alternates for which uncertainty remains concerning the accuracy of the reliability estimate. The preceding section considered those areas in which the computed probability of success was deemed to be biased. Appropriate adjustments were made. This section addresses areas in which the adjusted probabilities may well be in error; however, it is more an uncertainty than a bias, which we seek to identify.

In the solid-motor alternates, the influence of the motor-imposed environment on the spacecraft bus might be further considered. True, we have used an engineering solution (the use of a deployable heat shield) to cope with the most prominent effect of this environment—the impingement of intense radiant heat flux from the exhaust plume on the solar array panels, and have accepted the penalty in weight and in reliability resulting from this solution. But there are other influences of this environment. These include:

- The effect of the same radiant heat flux on the Planetary Science Package (PSP). The PSP is deployed outward early in the mission, and is outside the protection afforded by the shield. It would be difficult to locate it in a position during orbit insertion which would give protection from the heat flux without introducing considerable complexity in spacecraft geometry, deployment mechanization, center-of-mass control, and command structure. On the other hand, letting the PSP remain exposed may require a weight and reliability penalty in using thermal insulation to protect the PSP and its associated drive mechanism.

- The high- and medium-gain communications antennas must be articulated outward or forward to avoid conflicting with the heat shield deployment. If outward, the antenna drive design must be capable of coping with the adverse g-loading and the antenna is not fully protected from the radiant flux. In any event, the requirement for a preferred location is likely to limit or preclude the use of these antennas for communication to verify the spacecraft orientation for the maneuver.

- The low-gain antenna, deployed parallel to the roll axis in the liquid-engine alternates, would thus be located where it is extremely vulnerable to plume heating during orbit insertion. Several possible solutions exist: (1) Stow the low gain antenna in its launch position during orbit insertion, and redeploy afterward; (2) Conduct the entire mission with an antenna which does not project from the solar array plane (as in the TRW Task A design); (3) Abandon the low-gain antenna capability at orbit insertion (in this case it could be mounted in the nozzle, and blasted out at ignition); or (4) Have two low-gain antennas—one deployed for early-mission use, and one in the solar array plane for late-mission use—and a switch. It is clear that some penalty—weight, communications coverage, operational complexity, or reliability degradation—is incurred by each of these solutions.

In summary, there are aspects of the solid-motor environment which may not have been accommodated by the design to the extent it has been refined in this study, and which may well decrease confidence that the reliability analysis encompasses all the effects—environmental and operational—which may be induced.

A significant uncertainty in the reliability analyses is related to the possibility of stress corrosion of titanium propellant tanks containing N_2O_4 oxidizer. This uncertainty, which is the result of a paucity of test data, applies to all the liquid-engine alternates (LEM descent stage, Transtage, and custom liquid configurations) as well as to the combination solid-bipropellant alternate. This is the principal (but not the only) uncertainty associated with what is generally considered the "space storability" of liquid propellants. Again, should developmental testing confirm that this is an obstacle, there are alternate approaches, some of which can be expected to overcome the problem: use of additivies in the N_2O_4 to inhibit corrosion; use of aluminum rather than titanium tanks; lining the tanks. Again, various penalties may accrue. The LEM descent stage configuration may be less susceptible to stress corrosion effects, because of the reduction in tank pressure for the majority of the time of the mission.

Table A-4. Summary of Component Failure Rates
(Rates are expressed as failures per 10^6 hours)

Column	1	2	3
Data Source	FARADA, TRW	Martin	Grumman
Applicable to	LEMDS configuration, custom liquid configuration, and midcourse engine of combination solid-liquid configuration	Transtage configuration	LEM, Apollo Mission
Environmental Factors according to:	Table A-5	Table A-6	Table A-7
Component:			
Pressurant tank	.08	.07	.0029
Fill and drain valves	.123	5.7	.0025(a)
Cap	.6	.2	--
Filter	.196	.3	.0005
Solenoid valve	.56	11.0	.0137
Explosive valve (dual squib)	.09*	.09(b)	.0515
Relief valve	.67	5.7	.0054(c)
Burst disk	.6	.6	--
Thrust chamber, main	11.2*	15.0(d)	.152
Venturi or orifice	.15	--	--
Regulator	.671	2.03	.0264
Fitting	.02	.07	--
Accumulator	--	.07(b)	--
Pressure switch, per contact set	--	.50	--
Check valve	--	5.0	--
Quad check valve	--	--	.0117
Thrust chamber, ACS	--	1.5	--
Propellant tank	.18	.18	.0029
Bladder	.079	.60	--
Bipropellant valve	--	9.6	--
Pilot valve	--	3.2	--
Jet vane assembly	2.71	--	--
Bellows	2.23	--	--

Sources:

(*) Asterisked items are based on TRW test data

(a) Includes cap

(b) Martin value not available, TRW or FARADA data used

(c) Includes burst disk

(d) Martin value not available, estimated from TRW experience

Table A-5. Environmental K Factors

Applicable to LEMDS configuration (based on FARADA, TRW failure rate data), custom liquid configuration, and midcourse engine of Combination Solid-Liquid Configuration

Mission Phase	Environmental K Factor
Boost	1000
Interplanetary cruise	1
Midcourse corrections	50
Orbit insertion	100
Orbit cruise	1
Orbit trim	50

In addition, a factor of 0.5 is applied for pressurization-propellant feed system components (when not operating) to represent the fraction of failures which are considered critical to the mission.

Table A-6. Failure Rate Modifying Factors

Applicable to Transtage Configuration

	Environmental Factor (K)				
Equipment	Boost	Cruise	Main Engine Firing	ACS Firing	Nonoperating Modifier
A. Main engine components					
1. All components except engine components	70	1	50	3	0.1
2. Engine components	30	1	940	3	0.1
B. ACS					
1. All components except engine components	70	1	50	3	0.1
2. Engine components	770	1	145	3	0.1

In addition, a factor of 0.5 is applied for pressurization-propellant feed system components (when not operating) to represent the fraction of failures which are considered critical to the mission.

Table A-7. Environmental K Factors

Applicable to LEMDS configuration (based on Grumman-Apollo failure rate data)

Mission Phase	Environmental K Factor
Component operating	200
Component not operating, but under pressure	20
Component not operating, and not under pressure	0.1

Table A-8. Solid Motor Probability of Success (Combination Configuration)

Component	Number, n	Component Failure Rate, 10^{-6} Failures/Cycle	Number of Cycles Operation	Reliability
Igniter	1	2,237[a]	1	.9978
Motor	1	18,600[b]	1	.9814
Thrust vector control (liquid injection)	1	5,000[c]	1	.9950

Sources:

[a] Minuteman Stages 1, 2, 3 (753 tests—1 failure)
Nominal 50 per cent confidence = .002237

[b] AIAA Paper 65-165, "Malfunction Sensors for Large Solid Rocket Motors"

[c] JPL Memorandum 33-219, page 3, 10 May 1965

Table A-9. Solar Panel Shield Reliability (Combination Configuration)

Item	Failure Rate	Data Source	Operating Time-Hrs	Probability of Failure	Reliability
Squib	300×10^{-6}/Cycle	(a)	----	$.0^{3}30$	$.9^{3}7$
Actuation spring	$.012 \times 10^{-6}$/Hr.	(b)	4320	$.0^{4}52$	$.9^{4}48$
Cable	$.02 \times 10^{-6}$/Hr.	(c)	4320	$.0^{4}86$	$.9^{4}14$
Cable cutter	13800×10^{-6}/Cycle	(d)	----	.0138	.9862

Reliability of cable cutter assembly $= R_{squib} R_{cutter}$

$$= (.9^{3}7)(.9862)$$

$$= .9859$$

Probability of failure of cable cutter assembly $Q = 1 - R$

$$= 1 - .9859$$

$$= .0141$$

Reliability of dual cutter assemblies $= 1 - Q^2$

$$= 1 - (.0141)^2$$

$$= .9^{3}801$$

Reliability of shield operation $= (R_{redund.\ cutter})(R_{cable})(R_{spring})(R_{redund.\ cutter})$

$$\times (R_{cable}) = (.9^{3}801)(.9^{4}14)(.9^{4}48)(.9^{3}801)(.9^{4}14)$$

$$= .9^{3}336$$

Sources:

(a) TRW experience includes 2000 firings with zero failures which indicates a failure rate of 3×10^{-4} at 50 per cent confidence level.

(b) FARADA, page 2.374, source 138 (Martin) October 1963.

(c) FARADA, page 2.283, source 138 (Martin) October 1963.

(d) TRW development and qualification tests of bolt cutter assemblies disclosed 50 successful cuttings, without failure. This would indicate a failure rate of .0138/cycle with 50 per cent confidence level.

Table A-10. Monopropellant Liquid Midcourse Engine Probability of Sucess

(Combination Configuration)

Component	Number, n	Component Failure Rate per 10^6 hours, λ	Redundant	Equivalent Time, $\Sigma K_i t_i$, hours	Mission Reliability, R
Fill and hand valve	3	.00000007[a]	Yes - assumed capped	2340	.99999
Pressurant tank	2	.080[b]	No	2340	.99962
Propellant tank	2	.180[b]	No	2340	.99916
Filter	2	.196[b]	No	154[d]	.99994
Tank bladder	2	.079[b]	No	2340	.99964
Regulator quad	1	.671[b] for each regulator	Yes - within quad	--	.99999
Relief valve	2	.670[b]	No	234[c]	.99971
Connections (fittings)	24	.020[b]	No	2340	.99887
Squib valve complex (pressurant)	1	---	Yes, by use of solenoid valve	--	.99999
Squib valve complex (propellant)	1	---	Yes, by use of solenoid valve	--	.99770
Thrust section	1	Mission reliability for each item	Yes (two redundant pairs of thrusters)	--	
4 Injectors		.99937	--	--	
4 Thrust chamber assemblies		.99855	--	--	.99997
4 Thrust vector control assemblies		.99959	--	--	
2 Squib valve switching devices		.99967	--	--	
Total monopropellant system					.9946

Sources:

[a] FARADA - page 2.403 - Source 83 (Grumman 1961) indicates 6.15×10^{-6} for A/C. Lab basic equivalent is 1/50 or 123×10^{-9}. Capping valves with O-ring (failure rate of $.6 \times 10^{-6}$) decreases valve failure rate as follows: $(123 \times 10^{-9})(.6 \times 10^{-6}) = .01373$

[b] FARADA, TRW: Table A-4, column 1

[c] Since the relief valve incorporates a burst disk which will minimize the probability of valve seat leakage, the environmental factor is reduced one order of magnitude

[d] Lower factor due to omission of filter failure probability during coast phases

Table A-11. Bipropellant Liquid Midcourse Engine Probability of Success

(Alternate Combination Liquid-Solid Configuration)

Component	Number, n	Component Failure Rate per 10^6 hours, λ	Redundant	Equivalent Time, $\Sigma K_i t_i$ hours	Mission Reliability, R
Pressurant Subsystem					
Fill valve	1	.00000007[a]	Yes - assumed capped	2340	.99999
Relief valve	3	.670[b]	No	234[c]	.99956
Helium tank	2	.080[b]	No	2340	.99962
Squib valve complex	1	---	No	---	.99999
Filter	1	.196[b]	No	154[d]	.99997
Regulator quad	1	.671[b] for each regulator	Yes - within quad	---	.99999
Dual check	2	.828[c] for each check	Yes	2340	.99998
Connection	16	.020[b]	No	2340	.99926
Total pressurant subsystem					.9986
Propellant Subsystem					
Propellant tank	4	.180[b]	No	2340	.99728
Start bellows	2	2.3	No	2340	.98910
Fill valve	2	.00000007[a]	Yes - assumed Capped	2340	.99999
Squib valve complex	2	---	No	---	.99539
Filter	2	.196[b]	No	154[d]	.99994
Thruster	4	Mission reliability (each)	Two redundant pairs		
Injector		.99937			.99997
Thrust chamber assembly		.99855			
Gimbal assembly		.99959			
Squib valve (as switch device)	4	.99967			
Total propellant subsystem					.9818
Total bipropellant system					.9804

Sources:

[a] FARADA, page 2.403 - Source 83 (Grumman 1961) indicates 6.15×10^{-6} for aircraft. Basic lab equivalent is 1/50 or 123×10^{-9} Capping valve to reduce leakage potential would decrease fail rate to .000000073

[b] FARADA-TRW: Table A-4, column 1

[c] Environmental factor reduced one magnitude because relief valve incorporates a burst disk which minimizes probability of relief valve leakage

[d] Lower factor due to omission of filter failure probability during coast phases

Table A-12. LEM Descent Stage Configuration Probability of Success (Component Failure Rates per FARADA, TRW)

Component	Number, n	Component Failure Rate per 10^6 hours, λ	Redundant	Equivalent Time $\Sigma K_i t_i$ hours	Mission Reliability, R
Propulsion System					
Helium tank	2	.08[(a)]	No	2340	.99960
Filter	7	.196[(a)]	No	2340	.99651
Latching solenoid valve	2	.56[(a)]	Yes	487*	.99947
Squib valve	21	.09[(a)] per firing	In part	--*	.99987
Regulator	2	.671[(a)]	Yes	487*	.99937
Quad check valve	6	*	Within quad	*	.99989
Fill and drain valve	7	.10[(b)]	Yes - assumed capped	2340	.99725
Propellant tank	4	.18[(a)]	No	2340	.99832
Start tank - bellows	2	2.23[(a)]	No	2340	.9896
Relief valve	3	.4[(a)]	Yes - burst disk included	2340	.99700
Quad solenoid valve	4	*	Within quad	*	.99999
Orifice	4	.15[(a)]	No	2340	.99987
Cavitating venturi	2	.15[(a)]	No	2340	.99993
Thrust chamber assembly (includes injector, chamber, and nozzle)	1	11.2[(a)]	No	165[(c)]	.99815
Pintle actuator	1	3.6[(a)]	No	165[(c)]	.99940
Lines and fittings	20	.02[(a)]	No	2340	.99906
Total Propulsion System					.9736
Gimbals (2-axis)	1	3.9[(d)]	No	2340	.9940

*Effects of several failure modes and redundancy are considered.

Sources of component failure rate data:

(a) FARADA, TRW: Table A-4, column 1

(b) Adjusted from Table A-3, column 1 to account for redundancy of cap

(c) Adjusted to account for nonoperating periods

(d) FARADA, TRW: Composed of 2 gimbal actuators and 2 gimbal bearings. λ actuator - 1.8, λ gimbal - .1.

Table A-13 Transtage Configuration Probability of Success

Component	Number, n	Component Failure Rate per 10^6 hours, λ	Redundant	Equivalent Time $\Sigma K_i t_i$ hours	Mission Reliability, R
Main Engine Assembly					
Helium tank	2	.07 (a)	No	4500	.99937
Fill and drain valve	5	5.7 (a)	Yes - assumed capped	1.05*	.99999
Filter	2	.3	No	450 (b)	.99973
Quad solenoid valve	1	*	Within quad	*	.99998
Accumulator	1	.07	No	4500	.99937
Pressure switch	1	*	Assumed quad	*	.99998
Relief valve	1	*	Assumed redundant squibs	*	.99999
Check valve	4	5.0 (a)	Yes - 2 in series	*	.99941
Propellant tanks (includes check valves, traps, and screens)	2	5.18 (a)	No	590	.9941
Burst disk	2	.6 (a)	No	21	.99997
Lines and fittings	30	.02 (a)	No	2167	.9987
Bipropellant valve	2	4.8 (a)	No	390	.99353
Solenoid valve	2	11	No	390	.99149
Pilot valve	2	3.2 (a)	No	390	.99771
Thrust chamber assembly	2	15 (a)	No	105 (b)	.99685
Total Main Engine Assembly					.9704
Main Engine Gimbals (two-axis)	2	7.6 (c)	No	550 (b)	.9916
Attitude Control System					
Helium tank	1	.07 (a)	No	4500	.99968
Regulator and filter	2	*	Yes	*	.99999
Filter	1	.3 (a)	No	450 (b)	.99987
Check valve	4	5.0 (a)	2 each in series redundancy	*	.99941
Relief valve	2	5.7 (a)	No	215 (b)	.99754
Propellant tanks	2	.18 (a)	No	2340 (g)	.99916
Bellows for starting	2	2.1 (f)	No	2340 (g)	.9896
Fill valve	3	*	Capped	*	.99999
Manifold	2	2.9 (d)	No	550 (b)	.99849
Solenoid valve	8	11 (a)	No	240	.9791
Thrust chamber assembly (includes injector, chamber, and nozzle)	8	1.5 (a)	No	980*	.99883
Lines and fittings	33	.02 (a)	No	2167	.99858
Total Attitude Control System					.9608
Spacecraft Equipment					
Solar panel release	4	11.9 (e)	No	21	.9990

*Effects of redundancy are considered.

(a) Martin; Table A-4, column 2

(b) Adjusted for nonoperating portions of mission

(c) Includes 2 actuators and bearings each

(d) Martin Reliability Manual; two actuators and bearings per engine

(e) One actuator and squib valve assumed for each point

(f) TRW-FARADA: Table A-4, column 1 (no bellows on present Martin Transtage)

(g) Comparable LEM K factors used, to insure comparative influence

Table A-14. LEM Descent Propulsion Subsystem Reliability Estimates for Apollo Mission (Supplied by Grumman)

Equipment	Estimate
DESCENT PROPULSION SUBSYSTEM	.998830
Descent Engine, Variable Injector	.999688
Propellant Press and Feed Subsystem	.999142
Coupling, fuel, manual disconnect, fill and drain	.999995
Coupling, oxidizer, manual disconnect, fill and drain	.999995
Tank, helium, storage	.999994
Filter, helium, in-line non-bypass	.999999
Valve, helium, latching, solenoid operated	.999972
Valve, helium, explosive operated	.999895
Valve, helium, pressure reducing	.999946
Valve, helium, pressure relief and burst disk	.999989
Coupling, fuel, manual disconnect, fill and vent	.999995
Coupling, oxidizer, manual disconnect, fill and vent	.999995
Coupling, helium, manual, disconnect, fill and test point	.999995
Valve, helium, quad check	.999976
Filter, fuel, in-line, non-bypass	.999999
Filter, oxidizer, in-line, non-bypass	.999999

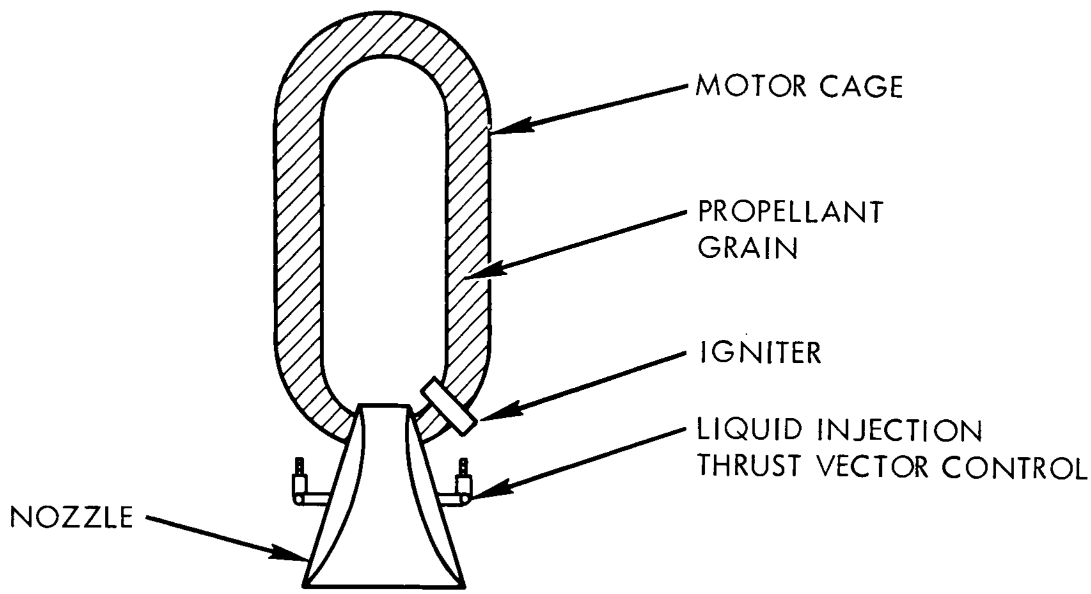


Figure A-1. Solid Propellant System

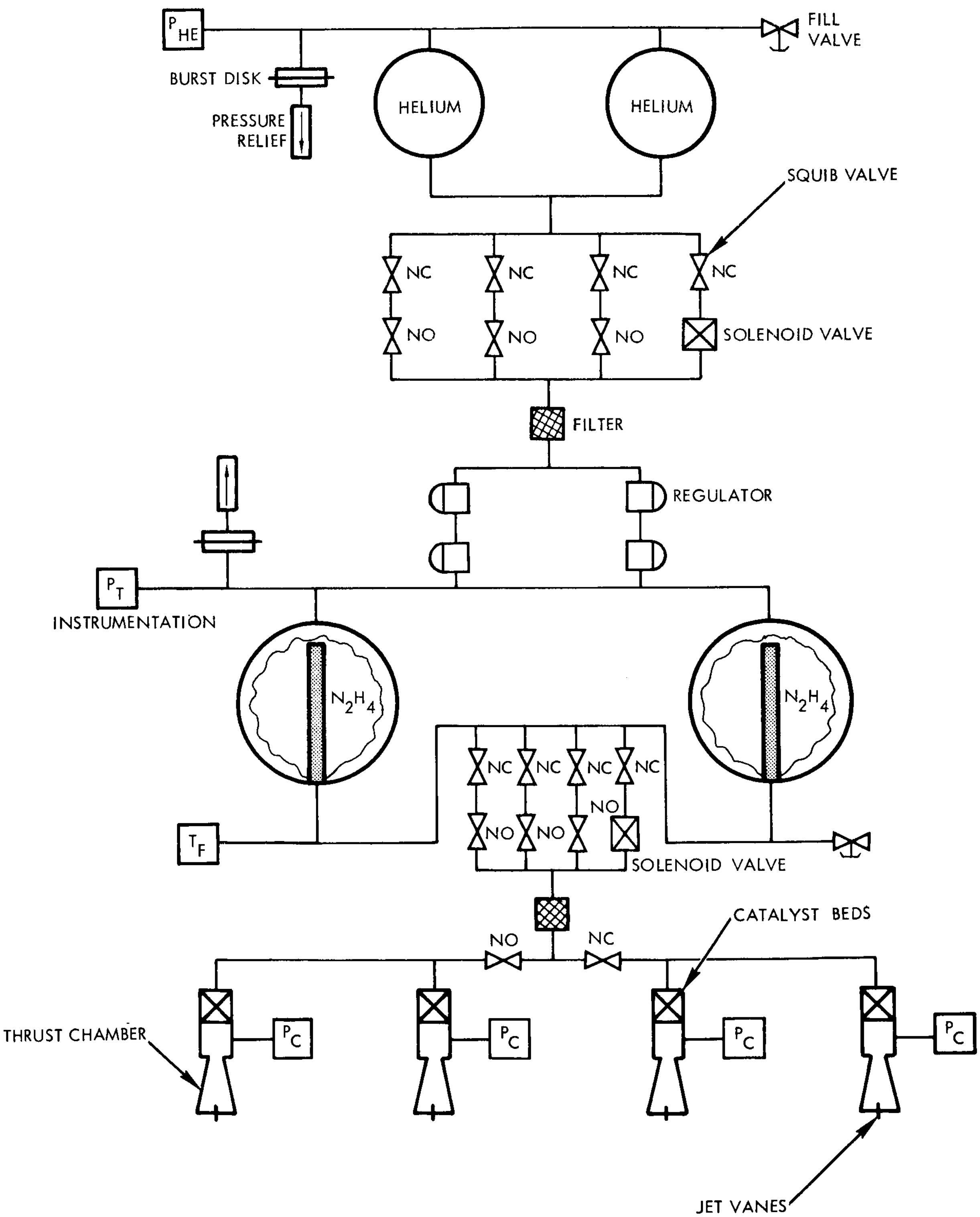


Figure A-2. Monopropellant System for Midcourse and Orbit Trim (Combination Solid-Liquid Configuration)

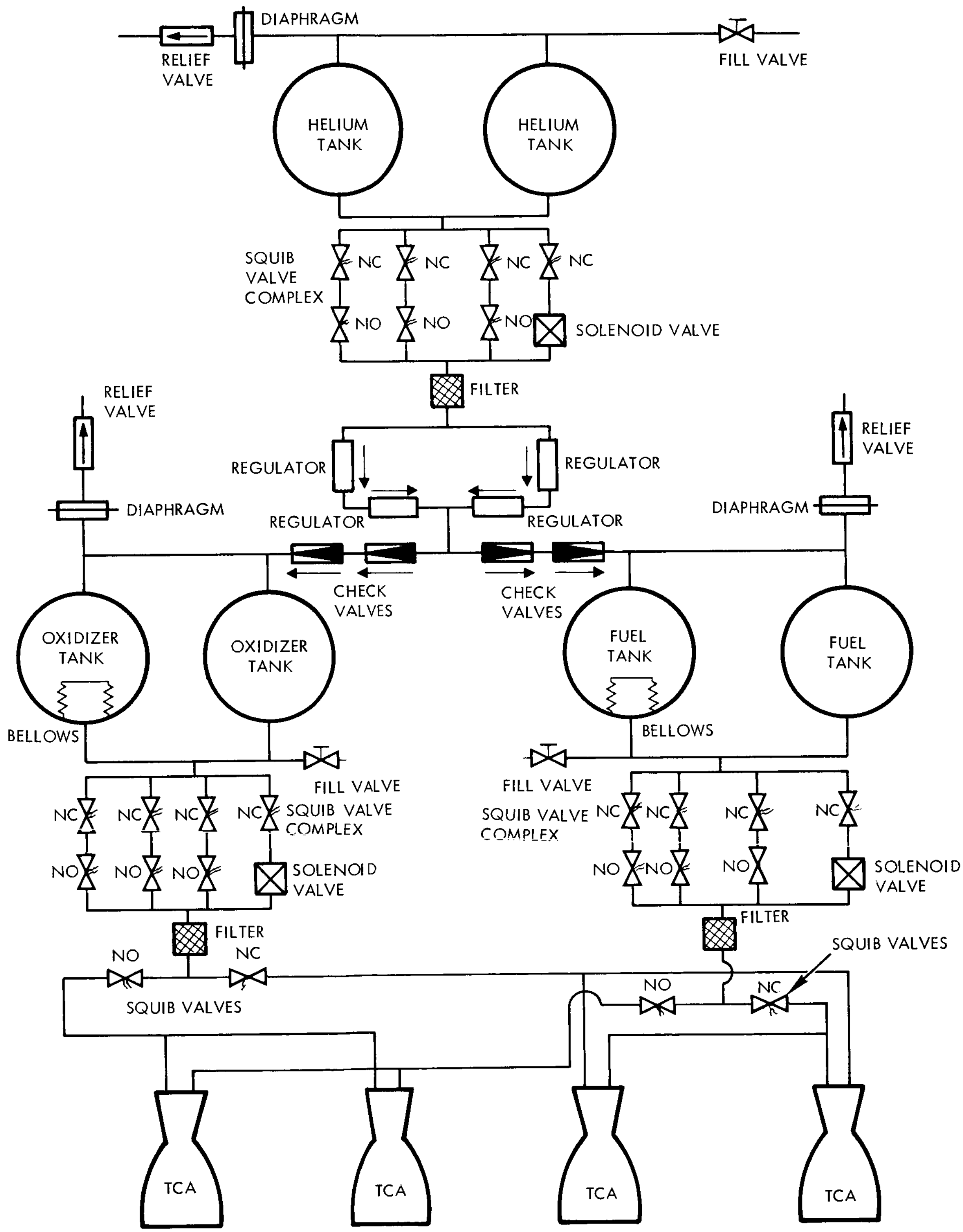


Figure A-3. Bipropellant System for Midcourse and Orbit Trim (Alternate Combination Solid-Liquid Configuration)

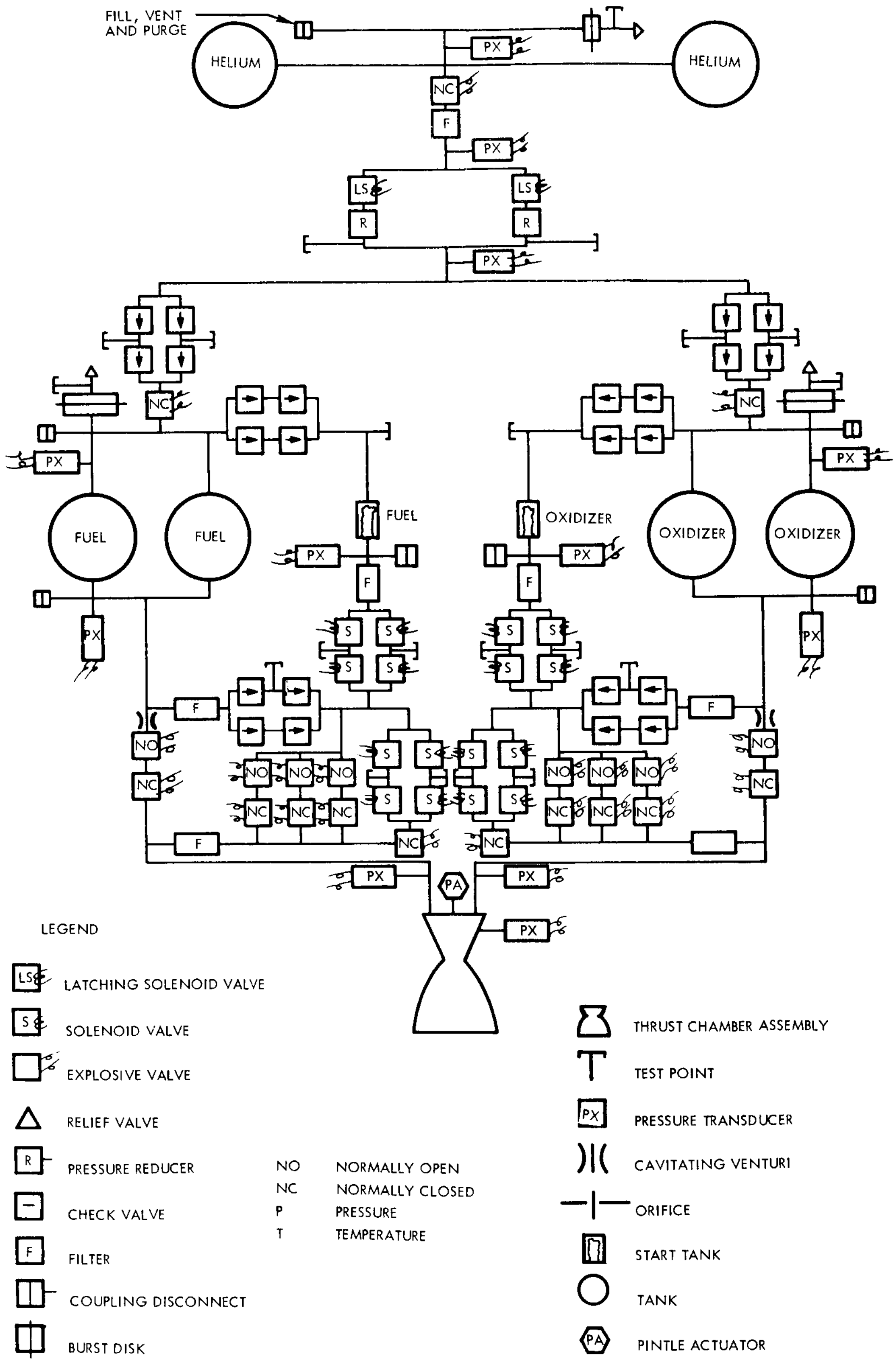


Figure A-4. LEM Descent Propulsion Stage (Modified for Voyager)

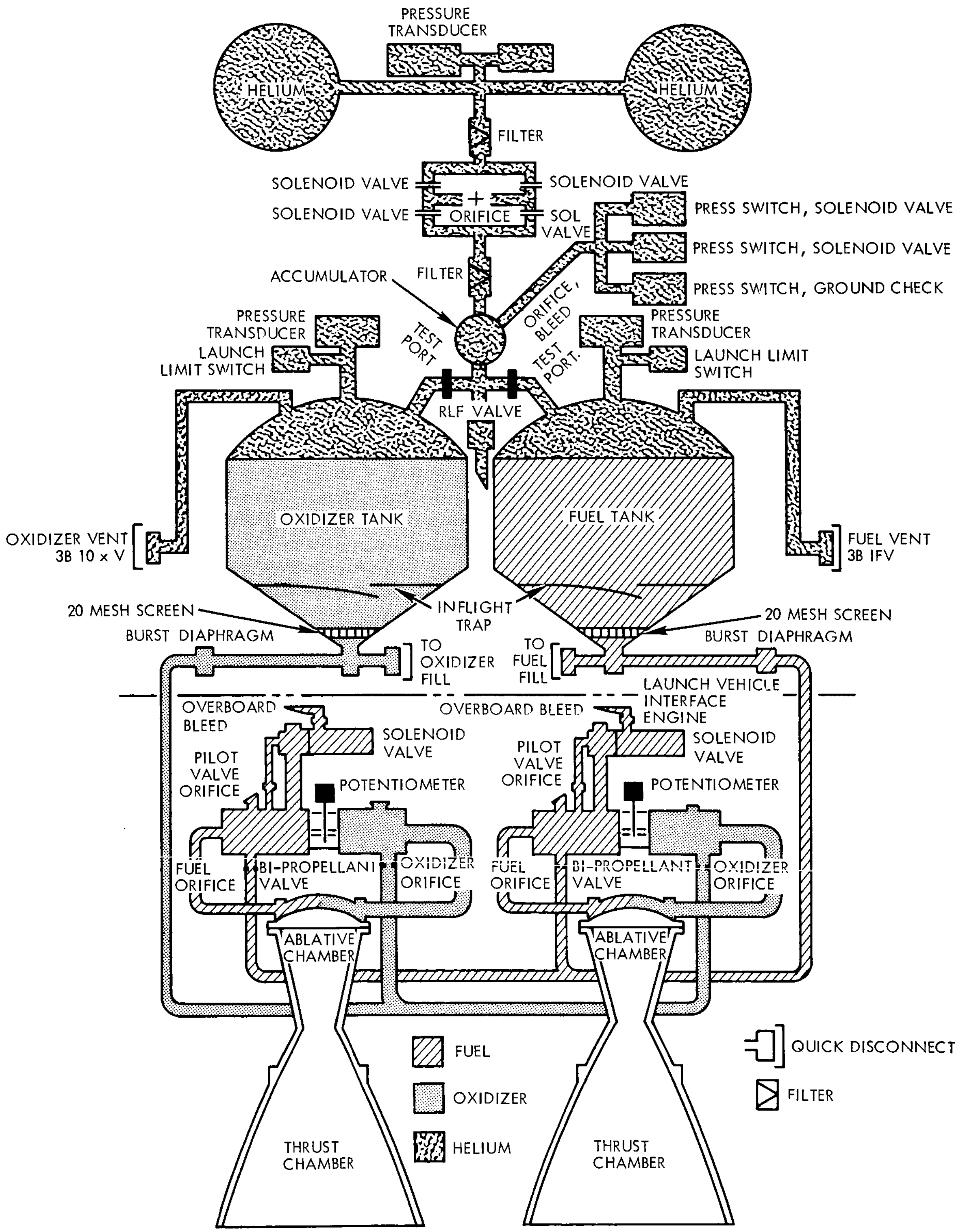


Figure A-5. Transtage Main Engine Propulsion System

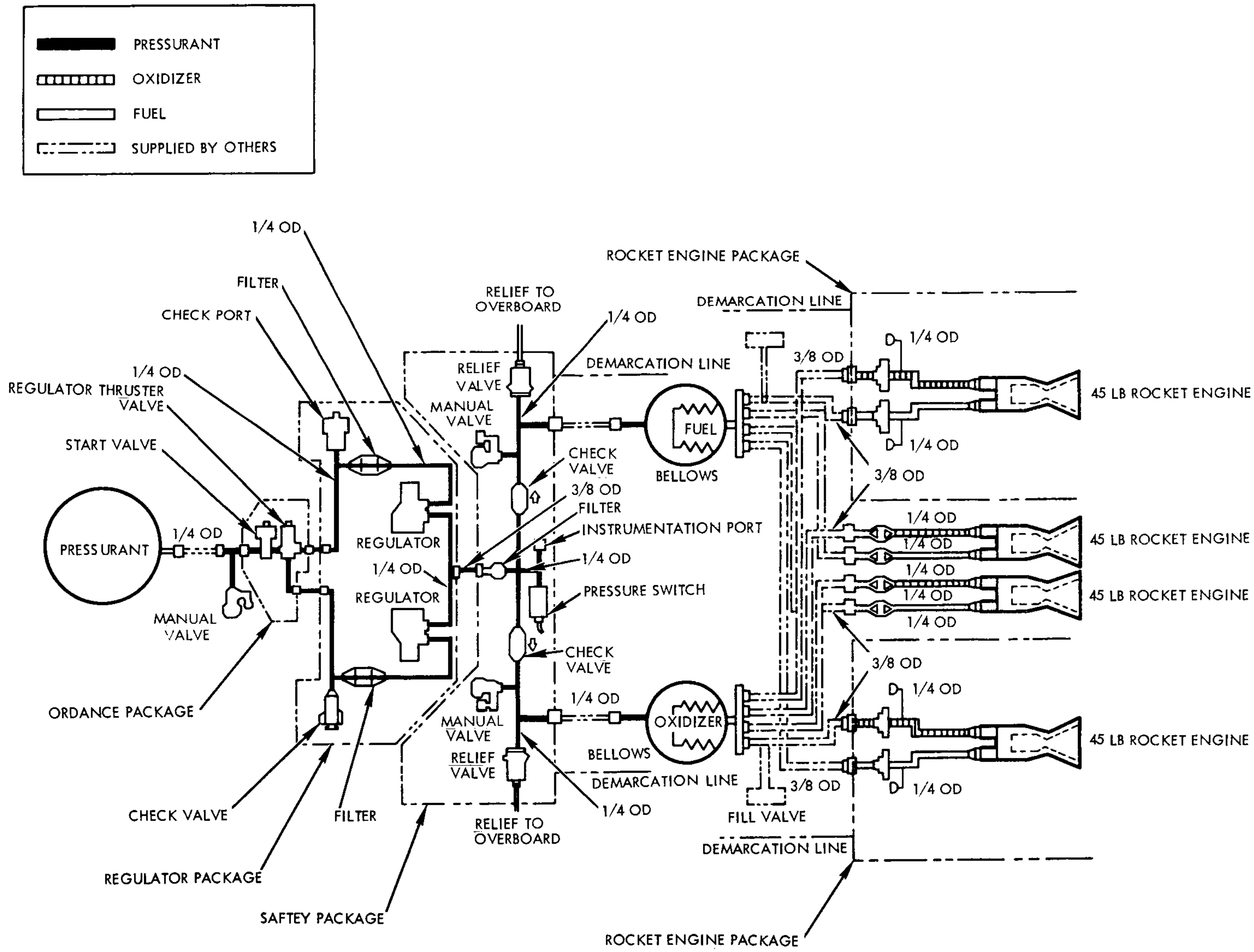


Figure A-6. Transtage Attitude Control System (ACS) Modified for Voyager Auxiliary Propulsion System

APPENDIX B

COST

1. GENERAL

One criterion for the selection of the preferred Voyager spacecraft propulsion system is cost. An initial evaluation has been made of the development and production costs of the four alternate propulsion systems to determine over-all effect on the Voyager spacecraft project costs. The results of the study indicate that the LEM descent stage configuration will be the least costly.

The cost analysis has been made based upon budget and planning factors generated by TRW from historical data on other spacecraft projects, quotation for similar items, and engineering judgments.

The approach to the study was to isolate items common to all configurations and concentrate only upon unique items. The electronic subsystems are similar for each configuration and were not considered. For the LEM descent stage, Transtage, and custom liquid configurations the assembly and test sequences are essentially the same and were, therefore, not costed. Although the assembly sequence of the combination solid-liquid configuration would differ slightly from that of liquid configurations, the cost difference was not considered to be a significant factor. The assembly, shipping, and handling equipment are different for each configuration but cost differences were not considered significant enough to be analyzed in detail. The cost analysis concentrated on an estimate of the development and production costs for the structural and mechanical sybsystems and the propulsion system. The results of the analysis are presented in Table B-1. Costs varied from a low of 55.2 million for the LEMDS configuration to a high of 79.6 million for the custom configuration. The solid configuration cost is 78.5 million, approximately the same as the custom. The Transtage configuration is 66.6 million. The overriding reason for the lower LEMDS cost is because of the lower development costs for adapting the LEM descent propulsion stage to the Voyager mission. Bus development costs are

Table B-1. Cost Comparison Voyager Configuration (Dollars in Millions)

	Solid	LEMDS	Transtage	Custom Liquid
Bus development	$11.5	$ 8.1	$10.1	$ 8.5
Propulsion system development	36.2	20.0	30.2	44.4
Total Development	$47.7	$28.1	$40.3	$52.9
Bus production	$12.0	$10.2	$10.1	$10.7
Propulsion system production	18.8	16.9	16.2	16.0
Total Production	$30.8	$27.1	$26.3	$26.7
Total	$78.5	$55.2	$66.6	$79.6

Note: Bus development and production costs are for structural and mechanical subsystems only. Costs include all direct labor, material, burden and fee.

essentially the same for all configurations. Production costs, except for the solid configuration, are almost equal.

The LEM descent propulsion system development costs are lower than the other three alternatives principally because it is a developed stage for long lifetime mission. The Transtage is a developed stage; however, it is used for short duration missions and requires major redevelopment to be used for the long duration Voyager mission. The solid and custom liquid configurations require major new structural development and the solid requires a major adaptation of an existing engine.

2. DEVELOPMENTAL COST CONSIDERATIONS

The developmental costs for the four configurations have been divided into the structural and mechanical development associated with the spacecraft bus and the structural, mechanical, and propulsion development required for the propulsion stage.

The spacecraft bus structure is essentially the same for all configurations. The development costs of the bus structure for the LEM descent stage are the lowest principally because of the well defined interfaces with the existing LEM descent stage; sizing of the LEM structure for the Saturn V, and no requirements for deployment of solar panels or heat shields. The custom liquid configuration bus structure is substantially the same as that for LEMDS. The Transtage configuration represents a substantially more complex design for the bus structure because the basic Transtage is sized for use with the Titan III and a new design structure to tie into the shroud is needed. Further, deployable solar panels are needed for Transtage which complicate development. For the solid configuration the complicating design requirements are the deployable heat shield and the heavier structure required for the solid motor.

The LEM descent propulsion stage requires a number of modifications for use with Voyager. Structural changes include lowering the engine, deleting the legs and outrigger, providing added micrometeoroid protection, and adding supports for reaction control elements. The LEMDS feed and pressurization system are modified to reduce propellant storage pressure, add a start system and add filters. Engine modifications include replacement of the radiation skirt with an ablative skirt, modification of the throttle linkage for two speeds, and removal of valves. Finally, new gimbal actuators must be developed.

Development costs of the Transtage propulsion system are 10.2 million higher than those of the LEM descent propulsion system. The Transtage development is considerably higher principally because Transtage was not designed as a long lifetime system. The long lifetime requirement necessitates a major change in the feed and pressurization systems including changing of tank volumes, brazing and welding of all plumbing joinings, addition of a start system, and modification of the altitude control system used for propellant settling. The engines are to be modified to use an ablative skirt and finally new gimbal actuators are to be designed.

The solid configuration requires the development of a new solid motor, a new midcourse engine, and a new structure. The solid motor is a scaled down version of the Minuteman Wing VI second stage motor. The changes to the motor, in addition to size scaling, involve a redesign of the case, removal of the roll control, revisions to the TVC pressure system, design of a new nozzle and nozzle extension, plus removal of external insulation. For midcourse correction a new hydrazine engine with feed and pressurization system must be developed. Finally, the structure for the propulsion system must be designed.

The most expensive alternative is the design of a new configuration based upon the LEM descent engine. For this alternative the same engine modifications as required for the LEMDS configuration must be made. Moreover an entirely new feed and pressurization system and a new structure must be designed and qualified.

In summary the development costs, particularly those of the propulsion systems, most strongly influence the cost differences between the various configurations. The LEMDS configuration is the least costly principally because it is a developed stage for space exploration. The Transtage is also a developed stage; however, major propulsion system changes are needed to adopt it to a long life span mission. The solid and custom liquid configurations represent major new development programs and costs are proportionately higher than either LEM or Transtage. The selected LEM configuration development costs are roughly one-half the custom liquid configuration, 60 percent of the solid configuration, and about 70 percent of the Transtage.

3. PRODUCTION COST CONSIDERATIONS

Production costs are based upon producing nine systems: three flight, a proof test model, a propulsion interaction model, and four type approval test units. For the solid configuration an additional 20 engines are added to the production figures to cover the needs for type approval testing and propulsion interaction tests. For the liquid configurations the propulsion system production costs include all structure, the thermal control, the engine (or engines) the fuel tanks and plumbing, the pressurization tanks and plumbing, and the integration and acceptance test of the propulsion system. Similarly in the solid configuration the propulsion system production costs include all structure, the solid motor, the midcourse engine and tankage, and the integration and acceptance test of the system plus 20 additional solid motors. The production costs of the bus structures include only the costs necessary to produce the structural assemblies and do not include any costs for integrating with other spacecraft subsystems.

Production costs for the liquid configurations are all essentially equal. Even though the custom configuration is a completely new design and has never been produced the expected propulsion system costs are equal to the other liquid alternates principally because a smaller and simpler structure would be utilized. Discounting the additional solid engines, the production costs of the entire solid configuration would be equal to the liquid alternates.

The production costs of the solid configuration bus structure is expected to be higher than the other alternates because of the considerably heavier trusses and supports. The bus structure for the liquid configurations are expected to be similar with the Transtage configuration slightly more expensive because of the requirement for deployable solar panels.

In summary, production costs for the alternate liquid configurations are essentially equal and are not a factor in configuration selection. The solid configuration is penalized in low production quantities because of the number of motors expended in test. Even considering the 1973 and 1975 mission the total solid configuration production costs will be greater than any of the liquids.

APPENDIX C
EXHAUST PLUME HEATING

1. SUMMARY

Plume heating effects caused by liquid and solid propulsion systems upon the Voyager flight spacecraft were examined. The results of the analyses establish the need for a radiation shield to protect the solar array for the solid propulsion systems. On the other hand, heating effects of the solar array and of other external equipment caused by exhaust plumes from the candidate liquid engines were determined to be negligible. The results of the study are not surprising since radiant heating from a solid particle plume is considerably greater than that from a gaseous plume.

The propulsion systems considered were:

- Solid Propellant Systems
  - a) Minuteman Wing VI 2nd stage, modified
  - b) Minuteman Wing V 2nd stage
- Liquid Propellant Systems
  - a) Titan IIIC Transtage
  - b) LEM descent stage

2. SOLID PROPELLANT

2.1 Results

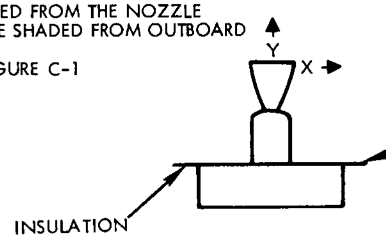
The results of the analysis for the solid propellant system are presented in Table C-1, and in Figure C-1. The results presented in Table C-1 are for the outboard edge of the array, which, as shown in Figure C-1, receives the greatest amount of incident heat flux and thus experiences the greatest temperature rise. Presented in Table C-1 are incident heat fluxes for several spacecraft configurations examined and the corresponding temperature rise of the solar array at the end of a 100-second deboost firing. As shown, for the configurations examined, the temperature rises at the end of firing are excessive and exceed the maximum allowable array temperature of 248°F. The need for a radiation shield is readily apparent. Therefore, it was conservatively

Table C-1. Solid Propellant Motor Plume Heating Results

PROPULSION SYSTEM	AXIAL DISTANCE (Y) FEET	RADIAL DISTANCE (X) FEET	INCIDENT HEAT = LUX, BTU/FT^2 HR	TEMPERATURE AT THE END OF 100 SEC FIRING	REQUIRED PLUME SHADING, * FEET
MODIFIED MM WING VI	-6.46	10	14600	1150	38
MODIFIED MM WING VI	-9.17	10	9660	900	34
MODIFIED MM WING VI**	-9.3	10	8300	800	25
WING VI	-12.7	10	4200	320	25
WING V (4-NOZZLE)	-6.46	10	9250	850	25

*THE LENGTH OF PLUME, AS MEASURED FROM THE NOZZLE EXIT DOWNSTREAM, WHICH MUST BE SHADED FROM OUTBOARD EDGES OF THE ARRAY.
**DETAILED RESULTS PRESENTED IN FIGURE C-1

MAXIMUM ALLOWABLE SOLAR ARRAY TEMPERATURE = 248°F



α/ϵ = 1 FOR PLUME RADIATION
ϵ = 0.8 FOR SOLAR ARRAY
WC_p = 0.17 BTU/HR-°F FOR SOLAR ARRAY

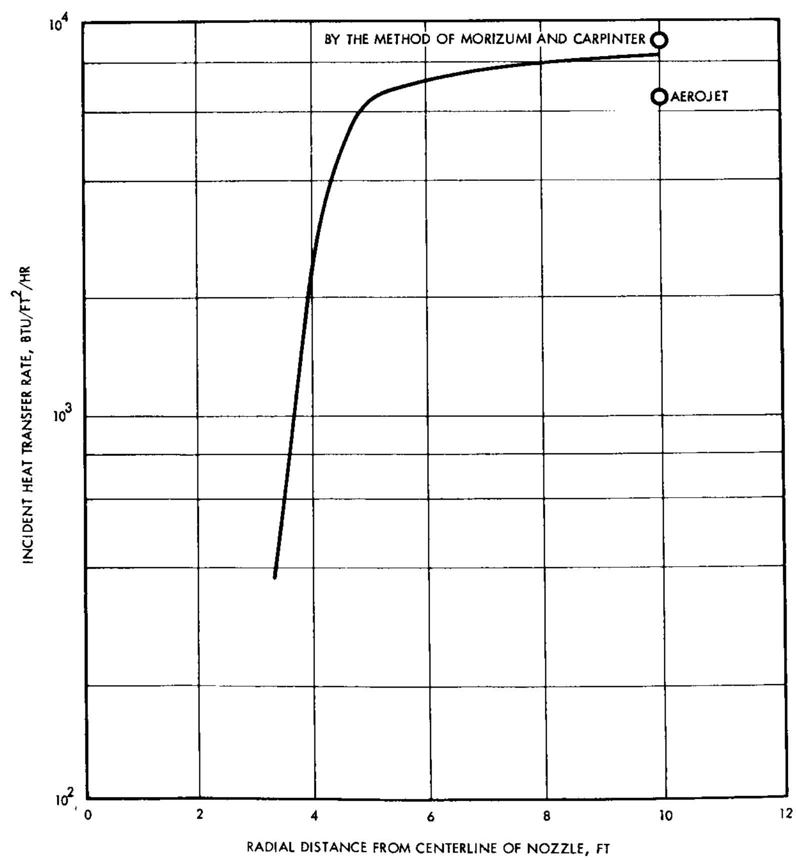


Figure C-1 Incident Heat Flux

determined what lengths of plume, as measured from the nozzle exits downstream, must be shaded from the outboard edges of the array to limit the incident flux to less than 400 Btu/hr ft^2.

2.2 Analysis

Two independent analytical techniques were used to evaluate incident radiant heat fluxes, both of which have been corroborated by test data. The first of these techniques is the method proposed by Morizumi and Carpenter.(1) The analysis treats radiation from a cloud of particles as that from an equivalent radiating surface. Thus, the problem is reduced to the determination of proper values of the apparent surface emissivity and the effective temperature. In defining the apparent emissivity, an analogy with neutron scattering for a cylindrical cloud is adopted which shows the apparent emissivity to be dependent on particle emissivity and cloud optical thickness. Since the plume is nonuniform in particle size, concentration and temperature, averaging techniques are used to define mean values of optical thickness and temperature. The particle flow-field (particle concentrations, temperatures, and trajectories) necessary to determine these two quantities was provided by a two-phase flow-field computer program.(2)

The second technique, which was used to evaluate the bulk of the data presented, is a simplified method of analysis in which the plume is assumed to be a cylindrical body of finite length and constant temperature. The effective plume dimensions and temperature are based on engine and nozzle parameters.* Comparison of the two methods of evaluating

(1) Morizumi, S. J. and H. J. Carpenter, "Thermal Radiation from the Exhaust Plume of an Aluminized Composite Propellant Rocket," Journal of Spacecraft, Vol. 1, No. 5, September-October, 1964.

(2) Nickerson, G. R. and J. R. Kliegel, "The Calculation of Supersonic Gas-Particle Flows in Axi-symmetric Nozzles by the Method of Characteristics," TRW Systems Report 6120-8345-MU000, May 1962.

* The method is documented in TRW Systems, Voyager Spacecraft, Phase IA, Part A Study Report, Vol. 5, Appendix I, p. D-30.

incident heat fluxes along with vendor data* is presented in Figure C-1. As shown, the two methods and vendor analysis agree quite favorably.

3. LIQUID PROPULSION SYSTEM

3.1 Results

Plume heating by connection and radiation were considered in the analysis for the liquid propellant systems. The LEM descent stage plume was analyzed in detail while the Transtage plume was analyzed by comparing engine parameters of the two systems and then judging the relative magnitudes of the environments. This approach was motivated by conservative results of the LEM descent stage which show negligible heating of the array, causing temperature rises of less than 5°F.

3.2 Analysis

Plume heating from a liquid engine is inherently less than that from a solid motor due to the lack of molten solid particles. This in conjunction with lower exhaust temperatures considerably reduces radiant heating. Convective heating caused by gaseous impingement is essentially the same for both liquid and solid propellant systems and is generally low.

3.2.1 Convective Heating

Flow properties within the LEM plume were generated using the method of characteristics. It was conservatively assumed that all kinetic energy is transformed into thermal energy as the expanding gases strike a surface. Transformation of energy was assumed to be indepenent of body shape, size, or surface inclination, but solely dependent upon location within the plume. This method lends itself to a conservative examination of the convective thermal environment. The equation for heat transfer used is:

$$\dot{q} = 1/2\, \dot{m}\, V^2 = 1/2\, \rho V^3$$

where

$\dot{m}$ = mass flow rate through a unit cross-section area

V = gas velocity

ρ = gas density

*Provided by Aerojet-General Corporation.

The convective heat transfer distribution for the LEM engine is presented in Figure C-2 for operation at the higher thrust level for orbit insertion. In nearly all configurations analyzed, the plume did not impinge upon the solar arrays. However, from Figure C-2 it can be seen that exhaust plume impingement would impose negligible heat transfer rates should this occur.

3.2.2 Radiation Heating

A prediction of the radiant heat flux of the LEM plume was made using data from the characteristic net analysis as a basis for a conservative estimate of the effective plume temperature. The apparent emissivity of the plume was taken to be 0.1. The simplified method of analysis previously described was then employed.

The results of analysis are presented in Figure C-3 for flat surfaces oriented normal to the plume centerline (facing both upstream and downstream) and for a surface facing the centerline. It is apparent that the heat transfer rates in the regions of the arrays are of a negligible level.

The solar array for the selected configuration is located approximately four feet from the nozzle exit. It can be seen in Figure C-2 that the plume would not impinge upon the solar array. This remains true even if the engine is gimballed hard-over. Even if the plume were to impinge on any component in the vicinity of the array, the heating rate would be much less than 10 Btu/ft^2hr, a negligible flux, causing a temperature rise of less than 5° F.

The omni-antenna and magnetometer locations for the configuration based on the LEM descent stage are given as a radial distance of 10 feet and an axial distance of 10 feet (on opposite sides of the spacecraft) as measured from the nozzle exit. Taking the worst case (engine gimballed hard-over and surface oriented toward the plume centerline), the convective and radiation heat transfer rates can be seen to be less than 100 $Btu/ft^2\,hr$ each (Figures C-2 and C-3). The total heat transfer rate (radiation, plus convection) of less than 200 Btu/ft^2hr does not create problems.

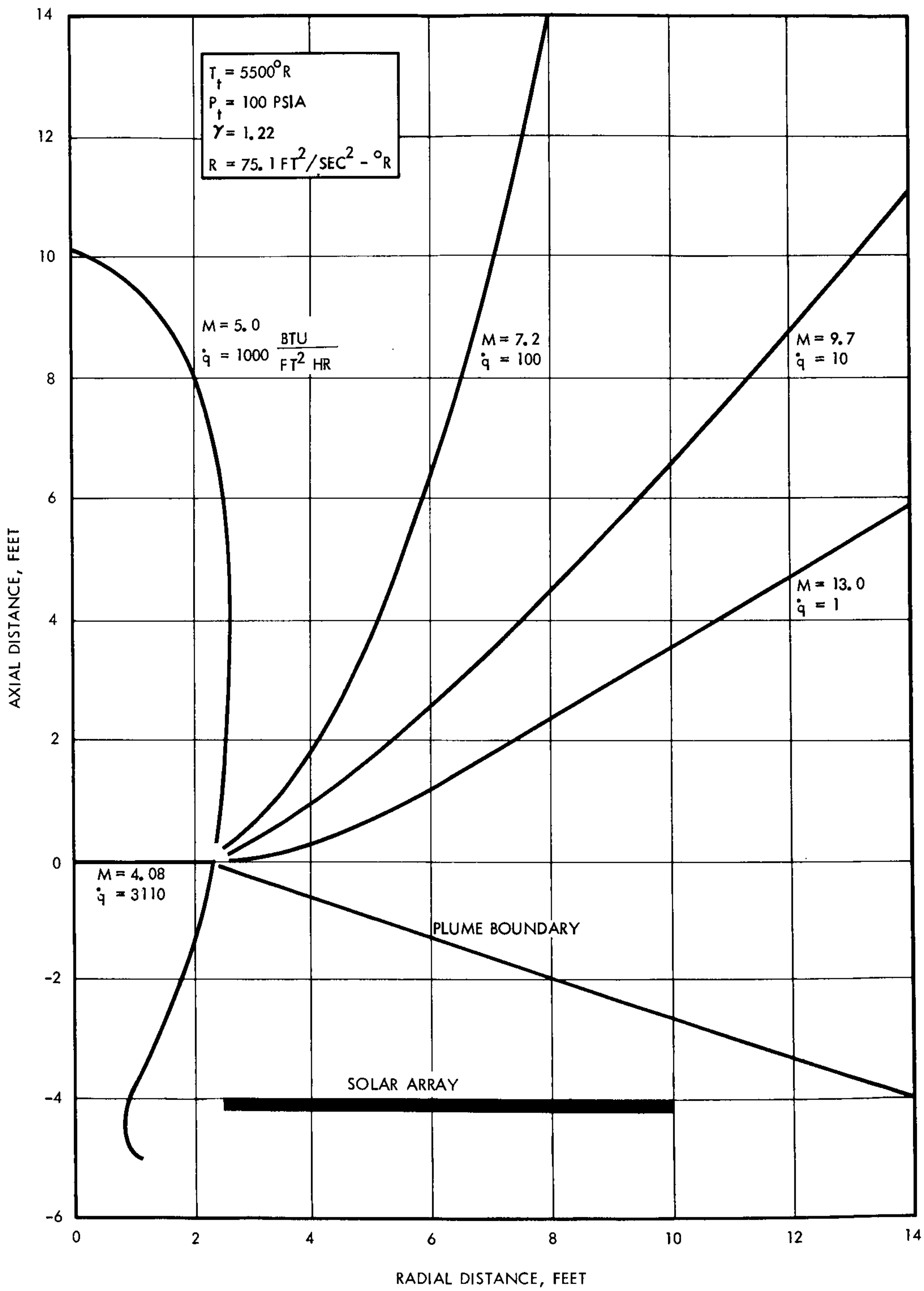


Figure C-2. Convective Heat Transfer of the LEM Engine

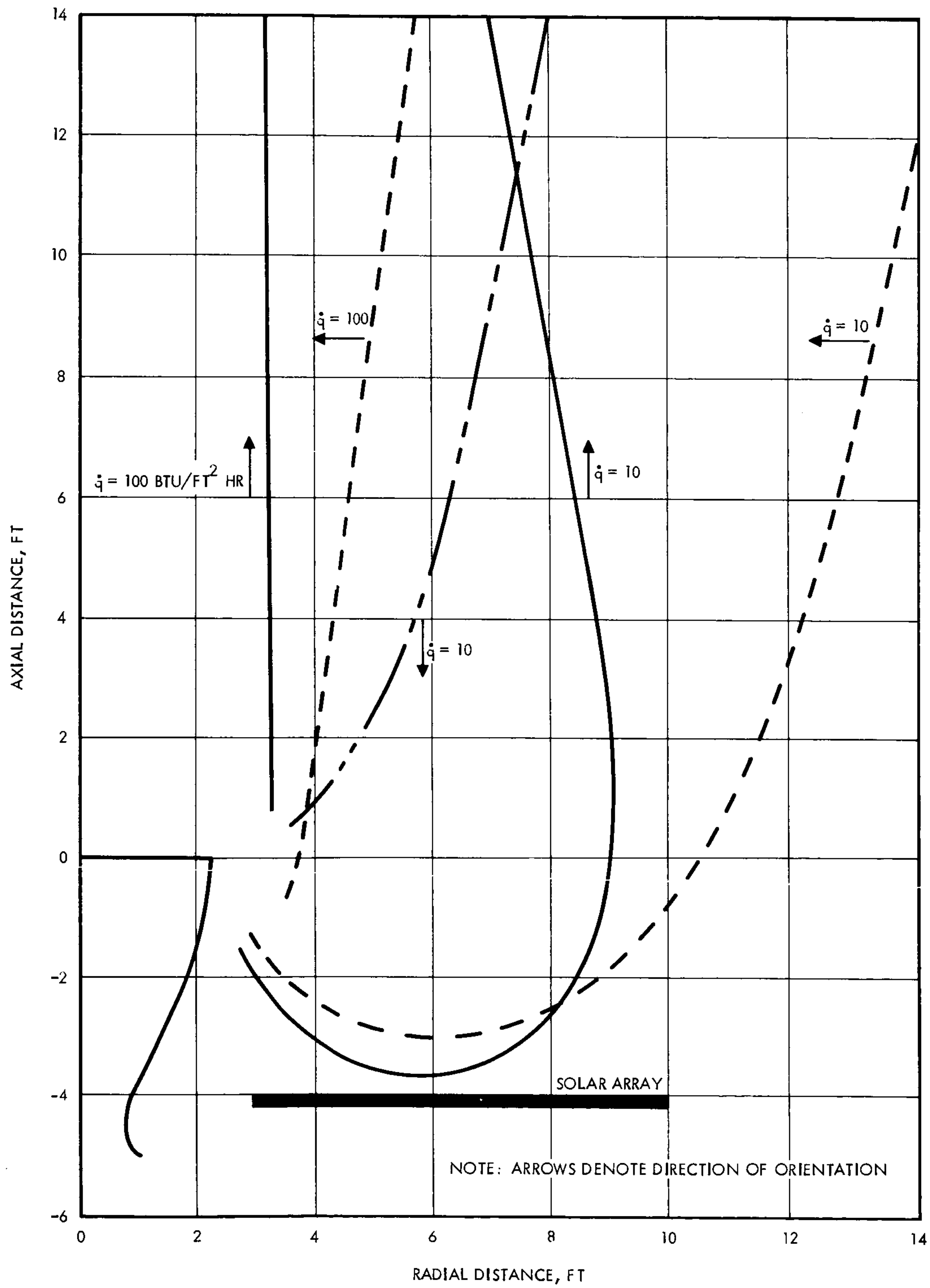


Figure C-3. Radiant Heat Transfer Distribution of the LEM Engine